SPACECRAFT DEPARTMENT

MISSILE AND SPACE DIVISION

GENERAL ELECTRIC

PROPELLANT STORABILITY IN SPACE

QUARTERLY RESEARCH CONTRACT STATUS REPORT NO. 1

15 MAY 1963 to 30 SEPTEMBER 1963

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UNITED STATES AIR FORCE
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TABLE OF CONTENTS

PROPELLANT STORABILITY IN SPACE

1.0 SUMMARY OF WORK ACCOMPLISHED 15 MAY 1963 TO 30 SEPTEMBER 1963

At the start of this report period, emphasis was placed upon the formulation of a thermal test plan. A preliminary version was reviewed with Edwards personnel on June 18, 1963. At this meeting several important decisions were reached regarding changes in the test program which differed from our original plan. These are summarized below:

1) In order to simulate an actual vehicle configuration as closely as possible, a simulated meteorite shield will be employed to enclose each of the tanks.

2) For the thermal interaction test between a cryogenic propellant and an earth storable, the use of two (2) four (4) foot diameter tanks rather than an eight (8) foot tank and a four (4) foot tank was agreed upon. This necessitates the procurement of an additional four (4) foot tank.

3) Because of physical limitations, the simulator tests will be performed with sun oriented tankage. However, the most probable mode of operation in space will be a randomly oriented tumbling vehicle. In order to obtain data for this type of vehicle, it was agreed to incorporate a rotating ($\pm 180^{\circ}$) tank into the pumpdown with the eight (8), four (4) and two (2) foot cryogenic tank test. Since this type of test is beyond the original scope of the contract, it has been included on a "best-effort" type basis. That is, every effort will be exerted to complete this portion of the test. However, should a malfunction occur, which might jeopardize the other tests, it will be shutdown such that the other tests may be completed.

4) The tests with cylindrical tankage will incorporate the same size (3' x 6') cylindrical tanks. They will be thermally isolated from one another, with one tank containing a cryogenic and the other containing an earth storable.

5) The test fluids selected were liquid nitrogen [illegible] methyl alcohol to simulate cryogenic and earth storable propellants, respectively.

6) Tests will be conducted at the Malta Test Station with LN_2, alcohol, nitrogen tetroxide, aerozene-50, diborane, and oxygen difluoride. These tests will be conducted in two (2) foot diameter spherical tanks in a transportable vacuum chamber. The tests with LN_2 and alcohol will assist in providing correlation data between the tests in the Solar Environmental Simulator and the tests at Malta.

With the basic ground rules established, a revised version of the program plan was prepared. Basically, two series of tests are to be performed in the Solar Environmental Simulator. The first test will be as depicted in Figure 1. Three separate tests will be conducted during this pumpdown, with the tests isolated from one another by means of nitrogen [illegible]. In one of the sectors a 3' x 6' cylindrical tank will be tested with liquid nitrogen. The other cylindrical tank of the same dimensions will use methly alcohol as the test fluid. The third test will consist of two spheres four (4) feet in diameter, placed side by side. With alcohol in one tank and liquid nitrogen in the second, this test will provide data on the thermal interactions between a cryogenic and an earth storable propellant combination.

Shown in Figure 2 is a schematic of the test arrangement for the second pumpdown. Four tests will be conducted using liquid nitrogen in each of the spheres. Three stationary spheres of 8, 4, and 2 feet diameter will be

tested to establish criteria for scaling. A second two feet diameter sphere will be mounted such that it will be rotated $\pm 180^{\circ}$ to determine gross effects of a tumbling vehicle. A secondary mirror system will be installed to reflect the sun onto the tank.

The final series of tests, as described previously, will be conducted at the Malta Test Station. For these tests, the heat loads obtained from the Solar Environmental Simulator experiment will be imposed upon the test tankage.

Having defined the basic test requirements, the design of the test tankage, meteorite shields, and support system was initiated. For the tanks which will contain alcohol, it was decided that an inner shell would be incorporated. This will result in a substantial reduction in the volume of the test fluid. Thus there will be a shorter time period required to reach thermal equilibrium. A gap of one inch between the inner and outer tank walls was selected to provide a reasonable manufacturing tolerance.

A particular area that has received major emphasis is that associated with the support of the tankage in the chamber. The basis of the experimental program is to duplicate, as closely as possible, an orbital configuration. It was established that heat leakage through the propellant tank supports must be given careful consideration. Unless advanced design techniques are used, the performance advantages of high energy cryogenic or semi-cryogenic propellants for missions requiring long term storage may be negated. As will be shown subsequently, an active type of tank support system shows considerable merit. In this sense, two support systems will be necessary. The larger system will accomodate the high "g" loadings encountered during boost and subsequent orbital firings. When the vehicle is in free flight, this support system is disengaged and the secondary system takes over. If the primary system were to remain in operation, excessive heat leaks would result.

In order to implement this concept in the simulator tests, reference is made to Figure 3. Shown here is a cross-sectional view through the longitudinal dimension of the cylindrical meteorite shield. The sketch pertains to the tanks containing liquid nitrogen. The tanks are supported by three support legs resting upon the base frame. These legs are hollow with liquid nitrogen circulated as shown. The purpose here is to minimize the heat input to the tank from these supports. In order to further reduce heat inputs from this source, the plastic thermal isolators are inserted as depicted in the sketch. The function of this system is to physically support the tanks in the chamber. However, the thermal isolation technique described above, practically eliminates heat transfer through the support elements. This then essentially simulates a zero "g" type of active support system. In order to simulate the support system during zero "g" orbital flight, four tension members attach from the meteorite shield to the tank. Further details are presented in a following section. Finally, the meteorite shields are supported from the base frame by low conductivity plastic support legs, minimizing the flow of heat between the meteorite shield and the base frame.

The method of support for the alcohol tanks is nearly identical to that used for the liquid nitrogen tanks. The exceptions are the absence of the cryogenically cooled legs, and the use of larger tension support members connecting the shield and tank. The larger tension members are permissable since it is expected that they will simulate a passive tank support system. Indications are that such a system should be acceptable for use with earth storable type propellants.

In order to accomodate the outlined Solar Environmental Simulator tests, effort has been directed towards the preparation of the test chamber. A base frame has been designed and is presently slated for manufacture. Orders are

in the process of being placed for a wide variety of equipment germane to this series of tests. Among these are the cryopanels which will separate the individual tests within each pumpdown, boil-off instrumentation, thermocouples, etc.

At the latter end of this quarter, planning efforts for the Malta tests were initiated, and a preliminary test procedure is in the process of compilation.

Additional details in each of the areas are discussed below.

2.0 Details of Effort in Major Areas

2.1 Thermal Analysis

2.1.1 Propellant Storage Program Plan

The initial program plan has been revised and updated and is presented here.

A. Test Orientation

1. <u>First Pumpdown in SES</u>

The tank orientation for this set-up is shown in Figure 1. During this test the tanks will be filled with liquid nitrogen or methyl alcohol as shown. Cryo-cooled panels separate the test tanks. The simulated sun will illuminate the test assembly from the top, and the earth and albedo fluxes will be introduced from a heated flat plate at the bottom. This set-up compromises the actual albedo flux which would be of solar wavelength instead of infrared. However, the yearly average absorbed albedo of a 66.5^{o} inclination angle orbit is approximately 4% of the total absorbed earth and albedo flux, when the external spacecraft surfaces have 0.21 solar absorptance and 0.85 emittance values. Thus this compromise is minor. The heat flux from this simulating plate should follow the orbital variation, but since the albedo fraction of the heat flux is the only part that varies and this component is small, constant temperature heater plates will be satisfactory. These plates should be insulated on the bottom and sized to radiate 70 watts per square foot. The sun should be cycled 66 minutes on and 28 minutes off throughout the test to simulate a yearly average orbit with solar intensity at 64% of one sun.

2. Second Pumpdown in SES

This test will be performed essentially as shown in Figure 2. The conditions and comments of A-1 are all applicable to this test series, except all tanks contain liquid nitrogen.

3. Malta Tests

Tests at this location will be performed with actual propellants in a two foot tank in a five foot vacuum chamber. Heater blankets will be placed around the outside of the cylindrical surface of the meteorite shield, and controlled to produce a temperature distribution around the shell approximating that measured during the SES test. A cylindrical liquid nitrogen cooled cryopanel will be placed around the cylindrical portion of the tank shield in order to provide a heat sink for desired temperature control. Detailed planning for these tests will commence soon.

B. Tank Design Thermal Restraints

1. Meteorite Shield

This member will be made from 0.1 inch aluminum and be cylindrical for all tanks. The ends of the cylindrical shields may be flat and of light gage and attached to the shell. An insulating blanket of 18 pairs of aluminum foil-tissue glass in about one-fourth inch thickness should be placed over the flat ends on the outside to provide a better simulation of an actual vehicle enclosure. The shield for the second four foot tank may be joined to the first one to form the dual tank interaction test, with the end plates and insulation removed at the joint. For this assembly, however, good thermal conductance between the two shields must be maintained.

The inside surface of the shields will be left as clean aluminum,

but the outside cylindrical surface of the shields will be coated with material having particular radiation absorptance to emittance ratios. This ratio for the cryogenic tanks will be as low as it appears reasonable to expect a space degraded coating to acquire. The ratio for the earth storable shields will be that which will maintain the shield at the average temperature desired for liquid storage, about 25°F. Initially the coating on the shield of the four foot cryogenic tank may need to be like the earth storables in order to produce the desired temperature in the adjoining liquid tank; then changed after the interaction test. The shields for the liquid nitrogen tanks should be painted with Pyromark white, manufactured by the Tempil Corporation. Extreme care must be taken after painting to keep this surface clean. According to previous tests this paint should have a solar absorptance of 0.21 and an emittance of 0.85 for an α/ϵ of 0.25. This ratio is probably representative of the degraded characteristic of $Z_nO_2/K_2\ S_iO_3/A\ell$, which is the coating that would be recommended for an actual vehicle. After 3100 hours of ultraviolet radiation at 1 AU intensity, the zinc oxide coating has a solar absorptance of 0.21 and an emittance of 0.90, for an α/ϵ of 0.23, according to Armour Research Foundation report ARF 3207-14 (Reference 1). The Pyromark white is not as stable under ultraviolet radiation, but this is not of particular importance, because the solar simulator does not include much energy of the damaging 2000 to 2600 angstroms wavelength.

The shields for the alcohol tanks may be painted with a combination of Pyromark white and Lowe Brothers black in a patchwork pattern to achieve the desired average α/ϵ ratio that will result in the 25° tank temperature, or with a single color paint if one with the desired characteristics is available. The preparation and grid size of the patchwork or the recommended single paint will be furnished at a later date.

The shield ends will need to have penetrations for the vent and fill lines and for the thermocouple instrumentation leads. A loose fit of the shield

around these penetrations will provide for venting the shields without the necessity for further vent holes.

The shields will be supported independently from the tanks. Wires representing the fixed part of the tank-to-shield support system will connect the shield and tank for the cryogenic systems, and rods will connect them in the earth storable systems, but will not be used during test for actual support. The shield to SES chamber support columns will include plastic separators near the shield end to minimize the thermal disturbance of the shield by the shield supports, and the columns will be wrapped with at least five layers of aluminized mylar insulation.

2. Cryogenic Tanks

(a) Insulation

All of the cryogenic tanks must be covered with three inches (about 200 layers) of super insulation. No special coating is required on the outside layer of the insulation.

The apparent conductivity of Linde SI 62 is given on Figure 4 as functions of the inside and outside insulation temperatures. Figures 5 through 10 convert this conductivity information to heat leak rates for different sized spheres. All of these curves are based on "theoretical" conductivity values and do not include any degrading factor for joints, discontinuities, and penetrations. A degrading factor of 1.1 to 1.2 is recommended by Linde.

(b) Supports

The LN_2 tanks will be supported in the test chamber by rigid, cryo-cooled legs, designed to transfer a minimum of heat through the supports to the tank. The supports will be hollow tubes, nearly full of LN_2, with a liquid-gas interface near the tank end of the tube. The legs should

include a plastic heat block between the tank and cooling section to further limit thermal conductance effects. Two hundred layers of aluminized mylar should be wrapped around each leg and interleafed with the tank insulation.

The best tank support system for an actual vehicle is believed to be a combination of fixed tension wires and movable support jaws, according to Section 2.1.2, Thermal Comparison of Tank Support Systems. The wires support the tank during the non-active portion of the mission (0.02g acceleration in any direction), and the moving jaws engage tank mounted lugs to support the tank during periods of higher acceleration loading. In an actual vehicle the jaws would be mounted on structure that is assumed to be a part of the micro-meteorite shield. As the jaws would be disengaged for most of the vehicle mission, that is the situation that will be duplicated during the SES tests. Therefore, it will not be necessary for the jaws to be included in this test, as a corresponding radiation connection between the shield and the tank lugs is already present. An illustrative jaw and lug design is shown as Figure 11. A low thermal conductivity titanium alloy could be used for the wires, with the gage size chosen according to the stress requirements. For these tests, however, stainless steel wires will be satisfactory.

The cryogenic test tanks do not need an inner tank shell for thermal reasons. Since the weight of full tanks can be accomodated by the SES chamber, the eight foot sphere, one four foot sphere and one cylindrical tank need have only one wall. An inner shell in the two foot tanks is desirable to reduce the volume and cost of actual propellant.

Each of the cryogenic tanks must include a fill and a vent line of the following sizes: 8' sphere - 1"; 4' sphere and 3' x 6' cylinder - 1/2"; 2' sphere - 3/8". These lines will be concentric from inside the tank to within six inches of the shield end penetration to reduce the heat leak into the tank through the lines. Inside the tanks the fill line will extend to the bottom and the vent line will end on the tank vertical centerline a small distance from the top of the tank wall. With this arrangement the tanks can be drained through the fill line, and any gas leakage past the fill line valve will not appreciably effect the tank boil-off measurement. Each of the fill lines must have a gas tight, remotely operable valve in the line immediately outside the meteorite shield penetration. The lines should be insulated for their exposed length both inside and outside the meteorite shield. The following sized strip heater should be wrapped around the fill and dump line just inside the shield end penetration:

2 foot sphere	1" x 2"
4 foot sphere	1" x 4"
8 foot sphere	1" x 10"
Cylinder	1" x 4"

These heaters will be used to raise the fill line temperature at the shield penetration to a temperature representative of the engine to which this line would be attached. The engine temperature will be considered to be the average temperature of the meteorite shield circumference.

One two foot tank will be rotated during test. Rotation through at least $\pm 180^{\circ}$ is desirable. To alleviate venting problems, the tank will be rotated about the vertical axis, with concentric fill and vent lines

attached at the top of the tank on this axis. The simulated solar flux will be reflected horizontally to this tank by a plane mirror mounted within the solar beam. Because the reflected flux will be at best about 85% of the intensity of the incident beam at the mirror, the shield of the rotating tank will be provided with a surface absorptance correspondingly increased.

The heat leak to the tank through the fill and vent lines will be different in the rotating test because of the different treatment of the lines inside the meteorite shield, resulting in higher losses for the rotating tank with the more closely coupled lines.

3. Earth-Storable Tanks

Alcohol is planned as the earth-storable test fluid. Because of the high heat capacity of the liquid, an inner shell will be needed in these tanks to reduce the length of time necessary to reach a steady operating test condition. A liquid annulus of one inch thickness or less is desirable. Use of alcohol will increase the rate of temperature change by more than a factor of two as compared to water. Ethyl alcohol has 91 percent of the heat capacity of methyl alcohol, so it would be slightly preferred.

The fill line should extend to the bottom of the tank. The vent line should come from the top. Both lines should be insulated for several feet distance within the meteorite shield. Valves may be installed in these lines to better simulate actual vehicle conditions, but it would be just as acceptable from a thermal standpoint to use valves outside the SES chamber.

A strip heater of about 300 watts capacity should be attached to the bottom of the tanks. The heater purpose will be discussed under Test Procedures.

Analysis has indicated the desirability of providing more thermal isolation between the meteorite shield and the tank wall than bare aluminum radiation interchange will give. This analysis shows that about 20 layers of super insulation in 0.3 inch thickness on the tank wall will limit the circumferential temperature difference to about 15°F.

The support legs for these tanks do not require special cooling, but they must incorporate a plastic heat block. Contrary to the requirements of the cryogenic tanks, the earth-storable ones could use a hard mount between the tank and the vehicle structure. A suitable mount for either the four foot diameter sphere or three foot by six foot cylinder would consist of six titanium rods of 0.66 inch diameter. Three hundred series stainless steel rods of 0.50 inch diameter may be substituted as the thermal equivalent. These rods could support the meteorite shields during test, thereby eliminating shield support legs, but current design uses the plastic shield support legs to also support the I.R. panel.

The hard mount for the non-cryogenic systems is thermally satisfactory for the following reasons:

(a) There is no net heat exchange between the shield and tank on a long time basis. Therefore mount conduction does not introduce a long term effect.

(b) The heat capacity of the filled tank is so large in relation to the thermal conductance between shield and tank for actual vehicle systems that the time constant for tank temperature response to shield temperature change is much longer than the cyclic period of shield temperature variation, even considering the long term effects due to orbit regression.

(c) The heat transfer rate to the tank through a rod connected to the warm side of the shield, or the rate from the tank through a rod connected to the cold side of the shield is not large enough to cause appreciable perturbations in local tank temperature.

C. Instrumentation

1. Boil-Off Measurement

Measurement of the rate of boil-off is planned for determination of heat flow into the cryogenic tanks. It is planned that this measurement will be made at atmospheric pressure and temperature of the effluent gas, with the gas supplied from test through the vent line from the tanks. Time variation of atmospheric pressure will have to be considered since this pressure variation can influence the boiling point of the nitrogen. If the boiling point changes during test, then the measured boil-off rate may either be magnifying or minimizing the heat flow rate into the tank. For instance, a change of 0.14 mm Hg in tank pressure will change the LN_2 boiling temperature by $0.00294^{\circ}R$ at one atmosphere pressure level. Twenty BTU are required to produce this change in temperature in the eight foot diameter tank. Therefore, a change in pressure of 0.14 mm Hg would cause the tank liquid to absorb or give up 20 BTU before equilibrium would again be restored. That amount of heat is more than the expected hourly heat leak rate into the tank. The schematic of a method to maintain constant pressure described in A. D. Little Report, "The Performance of a Double-Guarded Cold Plate Thermal Conductivity Apparatus", by I.A. Black and P.E. Glaser (Reference 2) is shown in Figure 12. This system is reported to hold a discharging gas pressure to ± 0.1 mm Hg of an absolute value.

The following gas flow rates represent the minimum and maximum amounts of boil-off expected from the cryogenic tanks during the periods when flow rate monitoring will be required. During fill and early stabilization periods, the flow rates are expected to exceed these values, but measurement will not be required.

Tank Size	Minimum	Maximum
	Cubic foot/hr. (NTP)	
2 foot sphere	0.3	0.8
4 foot sphere	0.6	1.3
8 foot sphere	1.6	2.0
3 x 6 foot cylinder	0.8	1.6

2. Temperature Instrumentation

(a) Meteorite Shield

Six 30 gage copper-constantan thermocouples will be mounted on the inside of the cylindrical shield surface in a hexagonal pattern with a flat at the top, plus one at the top. This pattern will be repeated each foot of shield length. Thus the two foot tank shield will have 21 thermocouples, 35 on a four foot tank shield, etc. In addition, five thermocouples will be placed on each end disc in a plus sign pattern.

(b) Cryogenic Tanks

The desire for temperature information from the cryogenic tanks must be compromised by the necessity for keeping extraneous heat leaks minimized. Leakage through thermocouple lead wires could add appreciably to this heat flow. Consequently it proposed that ten thermocouple pairs made from 36 gage copper-constantan wire be placed on the outside of the cryogenic tanks before application of the insulation. The insulated lead wires should be cemented to the tank shell with adhesive, and led out along the cryocooled legs, to which they must also be thermally attached. Thus,

separate penetrations are not required through the insulation for these wires, and heat flow to the tank through the wires is minimized.

Each vent line of the cryotanks will have a thermocouple attached about six inches from the point where the line emerges from the fill line. Filling of the tanks can then be confirmed by a sudden lowering of the temperature measured by this thermocouple. Two thermocouples should also be installed on the fill line, one a foot away from the tank and the other by the shield end penetration. Heat flow through the line to the tank can then be established, and the line end temperature can be controlled. One thermocouple should also be attached to each cryo-cooled leg near the vent location. Two forty gage copper constantan thermocouples should be attached to the outside layer of the insulation, with great care taken to cement the lead wires to the insulation for at least a distance of six inches from the junction.

(c) Earth Storable Tanks

The temperature and rate of temperature change of the earth storable tanks will be more critical from a testing standpoint than the temperature of the cryogenic tanks. Instrument lead heat leaks will be much less important in these tests because the temperature differences that cause such leaks are less. Consequently, the number of instrument points can be increased. Thirty-six gage copper constantan couples are recommended. The leads should be brought out around the legs. The four foot spherical tank should have nineteen couples on its outside surface, including one at the top, divided into three sets of six each. Each set lies in a plane that includes the vertical axis of the sphere, and the couples form a hexagon in this plane with a flat at the top. Each plane is 60 degrees

apart. The cylindrical tank will have twenty-eight 36 gage couples on it in the same pattern as on the meteorite shield, except that each circumferential row of couples is now 18 inches apart along the axis, beginning nine inches from one end of the cylinder. One couple should be attached to each line about one foot from the tank wall.

The legs of these tanks do not require special cooling, but they must incorporate a plastic heat block. One thermocouple should be attached to each leg at the outboard end of each plastic block.

(d) I.R. Heater Plates

Thermocouples should be attached to each of the simulated earth and albedo flux plates, according to the list under (e).

(e) Thermocouple Totals

1st Pumpdown

LN_2 cylindrical tank

Tank wall	10
Insulation	2
Lines	3
Legs	4
Shield	59
I.R. plate	6
	84

Alcohol cylindrical tank

Tank wall	28
Lines	2
Legs	4
Shields	59
I.R. plate	6
	99

Interaction Test		
LN_2 sphere		
Tank wall	10	
Insulation	2	
Lines	3	
Legs	4	
Alcohol sphere		
Tank wall	19	
Lines	2	
Legs	4	
Shield	73	
I.R. plate	8	
	125	
Total Test		308
2nd Pumpdown		
<u>2 foot stationary tank</u>		
Tank wall	10	
Insulation	2	
Lines	3	
Legs	4	
Shield	31	
I.R. plate	4	
	54	
<u>2 foot rotating tank</u>		
Tank wall	10	
Insulation	2	
Lines	3	
Shield	31	

<u>2 foot rotating tank</u> (cont.)

I.R. plate	4
	50

<u>4 foot sphere</u>

Tank wall	10
Insulation	2
Lines	3
Legs	4
Shield	45
I.R. plate	5
	69

<u>5 foot sphere</u>

Tank wall	10
Insulation	2
Lines	3
Legs	4
Shield	73
I.R. plate	9
	101
Total Test	274

3. <u>Pressure Measurement</u>

A transducer type pressure measuring instrument should be considered for each of the cryogenic tanks. All tanks will be operated with open vent lines, but knowledge of internal tank pressure would seem desirable for safety reasons, since vent line valve closing or line plugging are [illegible] pressure [illegible]. Perhaps this pressure gage could be a strain gage attached to the tank wall on the outside if cryogenic strain gages are available. High thermal resistance lead wires should be used with whatever instrument is used. For safety reasons,

a blowout disc will be built into each of the fill lines. This disc will be covered by the insulation, and the escaping gas will easily blow it away if sufficient pressure builds up in the tank.

Installation of a vacuum ion gage inside the meteorite shield was recommended by Dr. Kropschot of N.B.S. Such a gage should be installed on the inside of an end panel of the interaction test and a panel of the eight foot tank to measure the pressure to which the insulation is exposed.

D. Test Procedures

1. First Pump Down

This test consists of the interaction tests of two four foot spheres inside the same meteorite shield plus two cylindrical tanks. One sphere and one cylinder will be filled with liquid nitrogen and the others will be filled with alcohol. After test installation, instrumentation hook up, and test checkout, the operating procedure should be as follows:

(a) Begin pump down of chamber. At this point there is no liquid in the tanks.

(b) As soon as the chamber reaches the required vacuum level, admit nitrogen to the cryowalls of the chamber and to the cryopanels of the test set-up, and begin filling the alcohol tanks with 10°F alcohol. When the meteorite shields have reached predetermined temperature levels, the "sun" and the simulated earth and albedo flux should be turned on and the sun time cycles begun. This procedure begins to establish the proper temperature in the meteorite shields at the earliest possible time, and removes water vapor from the tank insulation.

(c) When the chamber has reached 10^{-4} Torr or less for a period of one hour, begin to fill the LN_2 tanks and tank cryolegs. Then shut down the diffusion pumps, and admit dry nitrogen gas to the chamber. Chamber pressure should be elevated to the 500 micron range, and held for sufficient time after the tank temperatures have dropped below -310°F for the outer insulation layer of the LN_2 tanks to begin to drop sharply in temperature. This procedure will decrease the thermal resistance of the super-insulation and reduce the time required to approach a quasi steady state thermal gradient in the insulation. An experiment will be run in the Thermal Laboratory to determine if 100 microns pressure will be sufficient instead of the 500 microns value.

(d) Continue filling the LN_2 tanks until either a sharp drop in the vent line temperature or some other means indicates that the tank is full. Shut the fill line valve at the tank at this point. Close the alcohol tank valves when liquid comes out the vent line, or when the tanks are full and cooled to 10°F if a circulatory system is used.

(e) When the drop in insulation temperature occurs, discontinue the dry nitrogen feed and pump down the chamber to 10^{-5} Torr or less. At this point, approximately the proper thermal gradient should be established in the insulation blanket, and the meteorite shields should have approached their long term cyclic temperature range values.

(f) Top off tanks, replacing the liquid boiled off since the initial fill. This may be necessary only for the two foot tanks.

(g) Continue operation of system, monitoring temperatures and boil-off rates, until steady operation has been reached. The boil-off rates, and perhaps tank temperatures, will show some fluctuation during a simulated orbit, and the meteorite shield will show a relatively large

temperature variation, but the temperatures and boil-off rates should show a repeatable cycle. This repeating cycle should begin to appear within a day.

(h) During the latter part of step (g), the integrated area average temperatures of the alcohol tanks and shields will be determined. If the alcohol tanks are cooler than the shields, as prechilling the alcohol should provide, power the tank heater to make the average temperature of the tank reach the average temperature of the shield. This is the situation that would eventually happen if the tanks were allowed to seek their own equilibrium temperature, but with the poor thermal coupling between shield and tank necessary to limit the liquid temperature range, the time constant is so large that exorbitant test times would be required. For the same reason, considerable care must be exercised during tank heater operation to insure that tank temperature overshoot does not occur.

(i) After the repeating temperature cycle has appeared, the test should be continued for two more days with continuous monitoring of boil-off rates and temperatures to determine that fully repeating values have been reached. At this point, with agreement of the thermal control engineer, normal shut down procedures may be initiated to stop the test.

(j) During step (i) the temperatures of the alcohol tanks and their shields will be continually monitored. These temperatures must be measured very precisely so that trends in direction of temperature changes can be determined. An instrument with the accuracy of L and N K3 potentiometer is recommended. The thermocouple system should be setup with sufficient flexibility so that manual reading with such an instrument can be made

when desired. Because the rates of temperature change will be low, this part of the test may need to run longer than the LN_2 tank test.

(k) During the test the measured data will be analyzed to determine the actual tank performance. After test this data will be consolidated, further analysis performed if necessary, and written up in a report on the determinations of this test.

2. Second Pump Down

This test consists of the three different sized stationary spheres filled with liquid nitrogen plus the rotating two foot sphere also with liquid nitrogen. All of the procedures given in D1 applicable to the LN_2 tanks will apply to this test. Tank rotation should begin during step (b).

E. Supplementary Laboratory Experiments

In addition to the main tests described above, several laboratory experiments are required to assure precise evaluation of thermal coating properties, to determine the degree of thermal matching between the actual and simulated sun, to check SES chamber pressures required to establish quickly the required thermal gradient in the insulation, and to evaluate the thermal conductance of the plastic separators in the tank support legs. These experiments will be performed prior to the propellant storage tests.

2.1.2 Thermal Comparison of Tank Support Systems

It has been established that heat leakage through the propellant tank supports must be given careful consideration. Unless advanced design techniques are applied here, the performance advantages of high energy cryogenic or semi-cryogenic propellants for missions requiring long-term storage may be negated. A survey of the literature, as well as personal contacts with several companies investigating this particular subject, has indicated that cryogenic tank support design has been clearly identified as a critical problem, but that no truly adequate solutions have been evaluated.

Indeed, as one begins to investigate the subject it is quite clear that it is worthy of a concentrated separate study to identify, analyze and test potential solutions. What is apparent at this time is that <u>advanced</u> propellants will require <u>advanced</u> design techniques. The time required to develop rocket hardware to use the new generation of space propellants is consistent with the time needed to develop the tankage and support systems.

For the purpose of conducting a realistic series of tests under the Propellant Storability Program, it is essential that the support system be thermally and structurally similar to that which might be ultimately used. Potential systems should not be eliminated at this time simply because they might be complex or require the development of a key component. The basis of selection of the support system will then be predicated upon three major requirements.

1) The system must be feasible (though not necessarily free from development problems).

2) The system must be thermally efficient.

3) The system must be capable of supporting the loads imposed during boost and during subsequent orbital maneuvers.

Two broad types of support systems have been considered here. One is a passive system which remains unchanged during the course of flight. The second type is considered an active system whereby some physical movement occurs to unload the system during zero "g" flight. There are a great many types of support systems that could be devised to meet the requirements. Presented here is a description of several systems. All of the structural/thermal designs and analyses discussed below and considered in the following sample calculations are based on the following conditions:

1) Eight (8) ft. diameter spherical cryogenic tank filled with OF_2.

2) Estimated acceleration during powered flight representing booster thrust and amplification of dynamic loads. 10 "g" axial--10 "g" lateral;

3) Acceleration during orbit due to centrifugal and aerodynamic drag forces (tumbling rate is 3 rpm with 6 ft. eccentricity; altitude is 260 n.mi.): 0.02 "g".

4) Acceleration during orbital firing: 1 - 2 "g";

5) Average micrometeorite shield temperature: $0^{o}F$. (α/ϵ = 0.3)

Passive Support Systems

1. Hard Mount - The hard mount shown in Figure 13 may be used as a basis for comparing the thermal efficiencies of other support systems. Although it is by far the poorest system thermally, it represents sound structural design. It provides a relatively high natural frequency for the tank/support system and as such could generally be considered to be least subject to high amplification of vibrations associated with the powered flight regime where maximum amplitudes usually occur in the low frequency portion of the spectrum.

2. Tension Members - Tension members, Figure 14, are better than hard mounts from the standpoint of heat leak; however, they are not structurally sound because of the inherent, relatively-low natural frequency of the type of tank/support system. Providing damping would alleviate this to some extent, but would lead to mechanical complexities and possibly introduce additional heat leaks.

3. Washers - Thermal resistance of a stack of stainless steel washers is directly proportional to the number of washers in the stack, and inversely proportional to the contact pressure between the interfaces to the 0.67 power (Reference 3). Thus, if high thermal performance is expected, the contact pressure must be minimized (0-20 psi) and many washers must be provided. Unfortunately, a stack of flat washers has no shear capacity. This problem can be alleviated if the individual washers are machined or embossed to provide interlocking (e.g. poker chips); however, such washers with adequate shear areas are relatively thick, so that for practical stack lengths, there would be too few washers to provide an effective barrier to heat flow.

A support system in which the shear of one stack of washers would be taken up in compression by another stack has been considered (See Figure 15). In order to insure a stable system, the washers must be placed under a considerable preload. Obviously, this compromises the thermal performance which could otherwise be attained.

A thermal analysis of the system illustrates that the full potential of washers cannot be realized when they have to be designed to provide the rugged structural requirements.

Active Support Systems

Generally, the tank support system will have to accomodate the extreme loads imposed during boost and the lesser loads imposed during orbital firings. It can be safely said that for any given mechanical support system the higher the load requirements, the more massive the supports, and the greater the associated heat leak penalties. By far, however, the greatest portion of the time is spent by the vehicle in an approximately zero "g" environment (accelerations associated with low tumbling rates and aerodynamic drag during orbit are quite small).

This strongly suggests the potential thermal superiority offered by a design which employs a primary support system which is engaged during boost and subsequent orbital firings and disengaged during orbiting, and an independent secondary support system, which is always engaged, that accomodates the near zero "g" requirements of orbiting.

The following support system appears best of those considered. It utilizes tension members for secondary support and the heat leaks they introduce are orders of magnitude less during orbit than for any known practical passive system with equivalent structural capabilities. Three active latch devices as shown schematically on Figure 11 are engaged during all ground and powered flight conditions but are disengaged during orbital operation. As the latch disengages, the superinsulation which had been compressed locally by the latch jaws expands and covers the lug attached to the tank. This feature reduces the rate of heat flow to the lug by radiation from the tank surroundings. A foamed insulator is attached to the end of the lug to extend the lug to the full depth of the insulation.

Table I illustrates the relative thermal efficiencies of the support systems just discussed in combination with three inch thick tank super-insulation whose theoretical performance has been degraded by a factor of 1.2. Boil-off rates in percentages per year are given for OF_2. These rates are based on an 8 ft. diameter tank and an average micrometeorite shield temperature of 0^oF. When considering boil-off rates, it is obvious that an active support system is the only type so far considered which will provide hard mount structural characteristics during launch and subsequent vehicle orbital firing but still permit low boil-off rates in the size tanks selected for this investigation. As the tank diameter is decreased, the need for an active support system is more pronounced.

To simulate any of the passive support systems, the system must be fabricated and installed in the test rig just as it would be in the spacecraft. The active system would require the secondary support wires and the tank mounted latching lugs. The structure mounted latching mechanisms are not, however, required for thermal simulation.

<u>Method of Analysis for Tank Support Systems Comparison</u>

The analysis presented here is directed towards computing the boil-off rate of OF_2 from an 8 ft. diameter spherical container covered with three (3) inches of super-insulation.

A. Evaluation of Parameters

1. <u>Orbit parameters</u>

a. i (inclination angle) = 66.5^o

b. h (orbit height) = 260 n.miles

c. e (orbit eccentricity) = 0.0

d. assumed vehicle configuration - cylinder

e. assumed vehicle attitude --

axis of cylinder is in the orbit plane and is normal to an earth-vehicle vector

f. attitude of orbit plane --

normal to sun-earth vector

2. Structural Design Considerations

a. dynamic loading during boost is assumed to be 10 "g" axial and 10 "g" lateral

b. Loading during orbit $\approx$ 0.0188 "g" (See Appendix I-1)

c. Design weight of tank completely full and with three inches of super insulation

$W_o \approx$ 27,500 lb. (see Appendix I-2)

3. Thermal Design Properties

a. apparent thermal conductivity of Linde SI 62 super insulation $k \approx 1.67 \times 10^{-5} \ \frac{\text{Btu}}{\text{hr. ft. }^{\circ}\text{F}}$ (see Appendix I-3)

b. properties of OF_2

a) ρ (density) = 95 $\frac{\text{lbm}}{\text{ft}^3}$

b) L (latent heat of vaporization @ 1 atmosphere) = 88.4 $\frac{\text{Btu}}{\text{lbm}}$

4. Structural Design Properties

a. material: Titanium 6AL 4V

σ_{TY} (tensile yield strength) = 120,000 psi

b. material: 301 stainless steel

σ_{CY} (compression yield strength) = 179,000 psi

c. material: 304 stainless steel

ρ (density) = 0.29 lb./in.^{3}

d. material: Linde SI 62 @ 70 layers/inch

ρ (density) = 6.2 lb./ft.^{3}

5. Average Micro-Meteorite Shield Temperature

$$\frac{1}{\pi}\left[S(1+aF_a)\alpha/\epsilon + \mathcal{E}\frac{F}{\epsilon}\right] = \sigma\bar{T}^4$$

S (solar constant) = 442 $\frac{\text{Btu}}{\text{hr. ft.}^2}$

a (albedo) $\approx$ 0.2

$\mathcal{E}$ (Earth emission) $\approx$ 88 $\frac{\text{Btu}}{\text{hr.ft.}^2}$

α/ϵ (solar absorptivity to emissivity ratio)

F_a (albedo factor) = 0.037

F_ϵ (earth factor) = 1.2

$\bar{T}$ (average micro meteorite shield temperature)

σ (Stephan Boltzmann constant)

coatings having an $\alpha/\epsilon \sim 0.3$ are available. Upon substitution, there results

$\bar{T} \approx 460^\circ R$ for $\alpha/\epsilon = 0.3$

B. Heat Leak Through Super Insulation

$$q = 4\pi \frac{(r_o + \tau)(r_o)}{\tau} K_a^1 \ (T_H - T_C)$$

r_o = 4 feet $\qquad T_H = 460^\circ R$

τ = 3 inches = .25 feet $\qquad T_C = 231^\circ R$

$K_a^1 = 2.00 \times 10^{-5} \ \frac{\text{Btu}}{\text{hr.ft.}^\circ R}$

$q = 3.92 \ \frac{\text{Btu}}{\text{hr.}}$

C. Hard Mount Analysis

1. Stress Analysis

Referring to Figure 13, the loading diagram is as shown, where the normal load F, is equal to 1/3 of the dynamic load at 10 g's, assuming three support points. Thus,

$$\frac{Mc}{I} + \frac{P}{A} \leq \frac{\sigma_{CY}}{S.F.}$$

where F (1/3 dynamic load) = 91,700 lb.

P (dynamic load) = 275,000 lb.

$A = \frac{\pi D^2}{4}$

I (moment of inertia) $= \frac{\pi D^4}{64}$

C (distance from neutral axis to outer fibre) = D/2

M (moment) = F $(0.5 + 0.75)$ in.lb.

S.F. (safety factor) = 1.25

solving by trial and error

D = 2.42 inches

L = 2.0 inches

2. <u>Heat Leak Analysis</u>

The structural support is assumed to be at the average micrometeorite shield temperature. Since there must be a very high contact pressure between the structure supports and the tank pads, the contact conductance of the interfaces has been experimentally evaluated as $\geqslant 200 \frac{\text{Btu}}{\text{hr.ft.}^2\,{}^\circ\text{F}}$

$$q \approx \frac{3\,(T_H - T_C)}{\frac{1}{hA_s} + \frac{X}{KA}} = 3720 \frac{\text{Btu}}{\text{hr.}}$$

$h \geqslant 200 \frac{\text{Btu}}{\text{hr.ft.}^2\,{}^\circ\text{F}}$

$A_s = \pi D(1.5\text{in.}) = \frac{\pi(2.42)(1.5)}{144} = 0.079 \text{ ft.}^2$

$X = 0.5 \text{ in.} = .0347 \text{ ft.}$

$K = 9 \frac{\text{Btu}}{\text{hr.ft.}^\circ\text{F}}$

$A = \frac{\pi D^2}{4} = \frac{\pi(2.42)^2}{4\,(144)} = .0319 \text{ ft.}^2$

Of course this simplified solution yields a higher rate of heat leak than would actually occur since heat drawn from the micrometeorite shield at the above rate would in turn reduce its temperature, thus

invalidating the assumption of $0^{\circ}F$ average shield temperature. By successive iterations one could arrive at a more accurate value for the heat leak; however, even if it were an order of magnitude lower, the hard mount support system would still be unacceptable from the standpoint of boil-off.

D. Tension Member Supports

1. Stress Analysis

This support system shown in Figure 14 consists of six rods at 45° to the micrometeorite shield axis and spaced 120° apart -- three on the top and three on the bottom. Assuming the rod material to be Titanium 6AL 4V, the maximum tension in any rod attributed to axial acceleration alone is

$$T_a = \frac{275{,}000}{3 \cos 45^{\circ}} = 130{,}000 \text{ lbs.}$$

In the worst case, two rods will take tension due to lateral acceleration

$$T_L = \frac{275{,}000}{2 \cos 45^{\circ}} = 195{,}000 \text{ lbs.}$$

The maximum tension in any rod is $T_a \quad T_L = 325{,}000$ lbs.

$$A = \frac{T_a \quad T_L}{\sigma_{TY}} = 2.71 \text{ in.}^2$$

2. Heat Leak Analysis

Assuming circular rods with an emissivity, $\epsilon = 0.06$, and thermal conductivity, $K = 4.0 \frac{\text{Btu}}{\text{hr. Ft.}^{\circ}\text{F}}$ and using equation #1 in appendix I-5 the heat input is found to be

$$q = 64.4 \frac{\text{Btu}}{\text{hr.}}$$

E. Washer Supports

1. Stress Analysis

As may be seen in Figure [illegible] the system consists of six (6) stacks of 302 stainless steel washers (0.0008 inches thick) mounted in compression along three (3) mutually orthogonal axes which intersect at the center of the tank.

Preload Required = 275,000 lbs.

Dynamic Loading = 275,000 lb.

F = 550,000 lb.

$$A = \frac{F}{\sigma_{CY}} = 3.07 \text{ in.}^2$$

2. Heat Leak Analysis

K (@ 89,500 psi) $\approx$ 3.03 $\frac{\text{Btu}}{\text{hr.ft.}^{\circ}\text{F}}$

from extrapolation of NBS data

L = 5 in.

$$q = 6 \frac{KA}{L} (T_H - T_C) = 213 \frac{\text{Btu}}{\text{hr.}}$$

F. Active Support System

1. Stress Analysis - Secondary Supports

The secondary wire supports are arranged in the same manner as the tension members of Figure 14. The wire material is Titanium [illegible]

These wires are designed to withstand only the accelerations during orbit $\approx$ 0.0188 [illegible] "weight" of tank = (0.0188)(27,500) = 517 lb., the wires should have a preload equal to the orbital "weight".

T = Total maximum tension = 1034 lb.

$$A = \frac{T}{\sigma_{TY}} = 0.00365 \text{ in.}^2$$

2. Heat Leak Analysis

Assuming wire of circular cross section with an emissivity, $\epsilon = 0.06$, and a thermal conductivity, $K = 4 \dfrac{Btu}{hr.ft.^{\circ}F}$, and using equation #1 in appendix I-5,

$$q = 0.56 \frac{Btu}{hr.}$$

There is an additional heat leak through the unlatched tank lugs by radiation. Assuming that these tank lugs must be equivalent to those used for a passive hard mount design, and that their radiating areas are equal.

$$\text{total } A_P = 3\left[\pi D\left(L + \frac{D}{4}\right)\right] \qquad \left\{\begin{matrix} D = 2.42 \text{ in.} \\ L = 2 \text{ in.} \end{matrix}\right\}$$

$$A_P = 59.4 \text{ in.}^2 \qquad \text{assume } \epsilon_P = 0.06$$

$$q_P \leq \epsilon_P \sigma A_P (T_H^4 - T_C^4) = 1.78 \frac{Btu}{hr.}$$

$$q_{W+P} = 2.34 \frac{Btu}{hr.}$$

G. Summary of Results

Neglecting the effects of plumbing, the heat leaks and boil-off rates are as tabulated in Table I.

H. Critique of Support System Utilizing Inertial Forces

In discussions concerning the relative merits of active and passive mounting systems for storable propellant tankage, one approach would use the inertia forces acting on the tanks due to steady state acceleration, to seat mounting lugs in receptacles on the structure after a certain amount of stretching had occurred in the primary support network of thin wires. To be workable, it appears that this method requires the steady state inertia forces seating the tank to be greater at any time, than the opposite acting component of the dynamic forces imposed on the tank during the powered portion of the flight regime.

Based on past experience and a check of a typical vehicle booster combination, this situation does not appear to exist, and consequently, it is felt that the above approach is not feasible.

As an example, steady state and dynamic data was obtained for a typical vehicle boosting an 18,500 # payload into a near earth orbit. The powered flight environment is as follows:

Steady State (due to rocket thrust)

Condition	Longitudinal G's (limit)
Solids Burnout	3.05 G's
1st Stage Burnout	4.00 G's
2nd Stage Burnout	2.17 G's
Trans Stage Burnout	.72 G's
Max. "Q"	1.09 G's + .565 G's lat.

Dynamic (Systems Qualification Levels) ultimate

Frequency	Longitudinal G's
5-10 cps	$\pm$ 2.25 G's (rms)
100-2000 cps	$\pm$ 3.0 G's (rms)
Random	
5-2000 cps	0.04 G^2/cps

It should be noted that the steady state "G" levels given at burnout of the various stages of the booster are maximum values, and will be much lower in the early stages of burning of each stage. Specifically, at lift off, steady state longitudinal acceleration will be quite low.

While the dynamic environment is given as a spectrum encompassing the entire flight regime, it is felt reasonable to assume that peak values can be experienced at any time during powered flight. Further, an estimate based on past experience with booster payload combinations of this type, indicates that amplification of dynamic loads at resonance could easily

increase input levels by a factor of 10 at the c.g. of the tank. Use of special high damping mounting structure in the vicinity of the tank might reduce this somewhat, but it is felt that the amplifications would not be likely to go below 5.

Therefore, assuming the critical natural frequency of the tank mounted on high damping structure to be below 100 cps, the most optimistic peak limit sinusoidal G's at the c.g. of the tank would be

$$\text{G Limit} = \frac{\pm 2.25}{\oplus 1.25} \times {}^{\boxplus}1.4 \times 5 = \pm \underline{\underline{12.6 \text{ G's}}} \text{ longitudinal}$$

$\oplus$Ultimate Load = 1.25 limit

$\boxplus$ Peak Load = rms $\sqrt{2}$ = rms (1.4)

It can be seen that even these optimistic values of dynamic load peaks far exceed any anticipated steady state loads due to booster thrust, and as a result, the propellant tank could be expected to oscillate in and out of contact with the mounting lugs with resulting high impact forces at the contact areas. The resulting structural loads on the tank would appear to be unacceptable. An additional consideration which tends to negate the thinking on this approach, is the effects of the sudden cutoff of thrust at the termination of each stage. Combined with the stretched condition of the primary wire support network, this would appear to produce a "sling shot" effect which would propel the tank forward out of the hard mounts with undesirable results, particularly, if ignition of the next stage occurred during the resulting oscillations. For these reasons, this mounting system is not considered to be practical.

I. Appendices

1. Orbital "g" Loading

assume: 1) 6 ft. eccentricity
2) 3 rpm tumbling rate

$$a = rw^2$$

$$= (6 \text{ ft.}) \left[\left(3 \frac{\text{rev}}{\text{min}}\right)\left(2 \frac{\pi \text{ rad}}{\text{rev}}\right)\left(\frac{\text{min}}{60 \text{ sec}}\right) \right]^2 = 0.592 \frac{\text{ft.}}{\text{sec.}^2}$$

$$= \left(0.592 \frac{\text{ft.}}{\text{sec.}^2}\right) \left(\frac{\text{"g"}}{32.2 \frac{\text{ft.}}{\text{sec.}^2}}\right) = 0.0188 \text{ g}$$

2. Design Weight of Tank System

$$W_o = W_T \text{ (tank)} + W_i \text{ (insulation)} + W_f \text{ (OF}_2) + W\text{(misc.)}$$

$$W_T = \frac{4}{3} \pi \left[r_o^3 - \left(r_o - \frac{\delta}{12}\right)^3 \right] \rho_T$$

$r_o = 4$ ft., $\delta = 3/16$ inch, ρ (304 S.S.) $= 0.29 \frac{\text{Lb.}}{\text{in.}^3}$

$$W_T = \frac{4}{3} \pi \left\{ (4 \text{ ft.})^3 - \left[4 \text{ ft.} - \left(\frac{3}{16} \text{ in.}\right)\left(\frac{\text{ft.}}{12 \text{ in.}}\right) \right]^3 \right\} \left(0.29 \frac{\text{lb.}}{\text{in.}^3}\right)\left(\frac{1728 \text{ in.}^3}{\text{ft.}^3}\right)$$

$$= 1568 \text{ lb.}$$

$$W_i = \frac{4}{3} \pi \left[\left(r_o + \frac{\tau}{12}\right) - r_o^3 \right] \rho_i$$

$\tau = 3$ inches, $\rho_i = 6.2 \frac{\text{lb.}}{\text{ft.}^3}$

$$W_i = \frac{4}{3} \pi \left\{ \left[4 \text{ ft.} + 3 \text{ in.} \left(\frac{\text{ft.}}{12 \text{ in.}}\right) \right]^3 - (4 \text{ ft.})^3 \right\} \left(6.2 \frac{\text{lb.}}{\text{ft.}^3}\right)$$

$$= 332 \text{ lb.}$$

$$W_f = \frac{4}{3} \pi (r_o - \delta)^3 \rho_f$$

$$\rho_f = 95 \frac{\text{lb.}}{\text{ft.}^3}$$

$$W_f = \frac{4}{3} \pi \left[4 \text{ ft.} - \left(\frac{3}{16} \text{ in.}\right)\left(\frac{\text{ft.}}{12 \text{ in.}}\right) \right]^3 \left(95 \frac{\text{lb.}}{\text{ft.}^3}\right)$$

$$= 25{,}170 \text{ lb.}$$

$$W_o = 1568 + 332 + 25{,}170 + W\text{(misc.)} \approx 27{,}500 \text{ lb.}$$

3. Apparent Thermal Conductivity of SI 62

$$K_a \text{ (70 layers/in.)} = 0.211 \times 10^{-6} \frac{(T_H^{3/2} - T_C)}{(T_H - T_C)} + 34.5 \times 10^{-6}$$ (cont. next line)

$$\sigma \frac{(T_H^4 - T_C^4)}{(T_H - T_C)}$$

T_H (hot side temperature) $\approx \bar{T}$ = 460^oR

T_C (cold side temperature) = 231^oR

σ (Stephan Boltzmann Constant) = $0.1713 \times 10^{-8} \dfrac{Btu}{hr.\ ft.^2\ {}^oR^4}$

$\bar{T}$ (average micro-meteorite shield temperature)

$K_a = 1.67 \times 10^{-5} \dfrac{Btu}{hr.\ ft.\ {}^oR}$

Linde has suggested using a de-rating factor of 1.2 in order to account for the discontinuities associated with an actual application.

$K_a^1 = 1.2\ K_a = 2.00 \times 10^{-5} \dfrac{Btu}{hr.\ ft.\ {}^oR}$

4. Conversion From Heat Leak to % Boil Off/year

$\gamma \dfrac{\sum_1 q}{W_f L}$ (100) = % Boil-Off = % B.O.

$\sum_1 q$ (summation of heat leaks $\dfrac{Btu}{hr.}$)

W_f (original OF_2 weight) = 25,170 lbm

L (latent heat of vaporization) = 88.4 $\dfrac{Btu}{lbm}$

B.O. (boil-off rate in % per year)

γ = 8760 hrs/yr

$0.394 \sum_1 q$ = B.O.

5. Radiant Fin Analysis (Rod or Wire Support)

eq. #1 $$|q| = \frac{KAM(T_H - T_C)}{\sin h\ (ML)} \quad ; \quad m = \sqrt{\frac{2\ \emptyset_r}{K} \quad \left(\frac{\pi}{A}\right)^{\frac{1}{2}}}$$

q (Btu/hr.) = heat leak per rod

K (Btu/hr. ft. ^oF) = thermal conductivity of rod

A (ft.^{2}) = cross sectional area of rod

L (ft.) = rod length

ϵ = emissivity of rod surface

$\sigma \left(\frac{Btu}{hr.\ ft.^2 - {}^oR}\right)$ = Stephan Boltzmann constant

$$\sigma (T_H^2 + T_C^2)(T_H + T_C) \leq \phi_r \leq \sigma 4 T_H^3$$

$$m = \left(\frac{\pi}{A}\right)^{\frac{1}{4}} \left(\frac{2\phi_r \epsilon}{K}\right)^{\frac{1}{2}}$$

In order to compute the limiting value of heat leak, $\phi_r = \sigma 4 T_H^3$ was used in the calculation. Equation #1 is the solution to the approximate linearized equation for a radiation fin of circular cross section for the boundary conditions stated below

$$\frac{d^2T}{dx^2} - m^2 (T - T_H) = 0$$

$$T(0) = T_C$$

$$T(L) = T_H$$

2.2 Structural Design and Hardware Procurement

During this reporting period, all test hardware was designed and procurement initiated for tanks, shields and support assemblies.

Tanks

Design of the test tankage involved two basic configurations: single-wall vessels for cryogenic fluids and a double-wall arrangement for earth-storable fluids. All tanks have identical vent and fill line systems, with the vent line inside the larger fill line as they enter the tank wall. Inside the tank the lines are separated with the vent line going to the top, and the fill to the bottom. Outside the tank, coaxially arranged flexible metal hose is used to allow bending the lines within the meteorite shield.

Several tapped bosses are provided on each tank exterior for the simulated meteorite shield to tank ties. In addition, a set of bosses

is provided to simulate an active tank to shield support for the cryogenic test vessels.

Externally applied silicone rubber heaters are provided on the 4' diameter and 3' diameter by 6' long double wall tanks, to decrease the time required to bring the earth storable fluids to equilibrium temperature.

The material chosen for the tanks was 304 stainless steel. This alloy is compatible with all the test fluids and with the Solar Environment Simulator test requirements. Material gauges chosen were based on the largest tank and used for all the tanks to facilitate ordering of raw materials.

Design conditions for all tanks are as follows:

Empty:

1. 5G vertical + 3G lateral + 3G side while on the supports (Road transport)
2. 3G lateral + 3G side in insulation wrapping configuration.
3. 3G vertical + 3G side on supports (handling condition).

Full:

1. 1G vertical on supports at operating temperatures.

Support Assemblies

All tanks are supported on hollow steel legs which are bolted to aluminum I-beam base frames. The hollow legs are provided with liquid N_2 ports for cooling purposes on all cryogenic fluid tests. Earth storable tests do not require these cooled support legs.

A glass cloth-epoxy resin laminated tube is inserted between the tanks and their support legs to further achieve thermal isolation between these elements. Selection of a material with low thermal conductivity

combined with adequate strength at cryogenic temperatures was made with the aid of tests at -300°F. The laminate mentioned above was selected and all tests have shown it to be suitable for the intended application.

Meteorite Shield Assemblies

The meteorite shields, designed to surround each tank have been standardized as flat-ended cylinders, ten inches greater in diameter and length than the respective tank diameters and lengths. The shields are aluminum shells 0.10 inch thick, with appropriate cutouts for the tank legs and vent lines. Joints along the length of the shields are provided to facilitate assembly around the tanks.

Except for the rotating two foot diameter tank, all shields are supported by legs which are attached to the tank support systems at their lower ends. These shield supports also serve as supports for the earth-radiation heater panels. Since the shield support legs are mainly laminated plastic tubes, they make excellent heat flow barriers to reduce heat conduction from the radiation heaters to shields. The shield for the rotating two foot diameter tank is supported by the wires connecting shield and tank.

2,3 Test Preparation and Operation

2.3.1 Test Preparation

A preliminary test plan has been issued which defines the detailed procedure to be followed in preparing the tanks, the auxiliary test equipment, and the SES facility. The plan is currently being modified to incorporate the latest data.

The base frame which supports the test tankage and auxiliary equipment, has been designed and is currently being fabricated. Aluminum "H" beams, 6" x 8", are being used with gusseted welded joints. The base frame will be painted black to minimize reflected radiation and cryogenically cooled with liquid nitrogen to minimize infra-red radiation to the test tank.

In conjunction with the test on the rotating two foot diameter tank, a tank rotator assembly has been designed. By incorporating flexible vent and fill lines, the tank may be rotated continuously through an angle of $\pm 180^{\circ}$ at a constant period of rotation of nine (9) minutes/cycle. In order to permit continuous venting of the tank, it is necessary to rotate the tank with the vent line remaining vertical. However, in order to maintain the same orientation of the sun with respect to the meteorite shields as in the fixed tank experiments, it has been necessary to provide a first surface mirror. This mirror is oriented at 45°, as shown in Figure 16, and it will permit the solar flux to maintain the desired geometrical relationship.

Detailed plans have been completed to prepare the tankage for testing, after the insulation has been applied. Among the tasks to be performed are the installation of vent and fill piping, fill shut-off valves, connection of the cryogenically cooled support system, installation of IR heater pads, and assembly of meteorite shield. Cryogenic panels have been designed to provide thermal isolation between tanks and fabrication will commence shortly. A thirty-six (36) inch penetration plate has been designed for the test chamber. This will provide for the entrance of liquid nitrogen lines to the tanks, cryopanels, support systems, etc.

The instrumentation for measuring propellant boil-off has been designed and is currently being ordered. A schematic is shown in Figure 17. The system will provide a constant boil-off pressure of 810 mm Hg to the test tankage with provision for adjustment to maintain this pressure constant to ± 0.1 mm. The actual metering will be performed by calibrated wet type positive displacement flow meters. The expected accuracy is approximately $\pm 0.5\%$.

The existing IR power control equipment and associated programmers are being installed in the SES facility. They will provide power to the tank IR pads, controlled to predetermined temperatures. Design and procurement of the alcohol fill and drain system is also underway. A schematic of the system is shown in Figure 18.

2.3.2 Test Operation

Initial planning for the detailed test operation sequence has started. The requirements specified by the insulation vendors, the thermal group, and the tank vendor are being constantly reviewed in terms of test facility requirements and capabilities to assure compatibility.

Figure 19 is a schematic of expected chamber performance during the tests. Pumpdown and warm-up time are estimated, based on past chamber operating experience and may vary somewhat depending on characteristics of the test objectives. Facility operating procedures are expected to be essentially the same for both Phase 1 and Phase 2 pumpdown, although the time required to temperature stabilize the alcohol-filled tanks used in Phase 1 pumpdown may be slightly longer than the time required to temperature stabilize for Phase 2 pumpdown, where all tanks are LN_2 filled.

A. Pumpdown

Pumpdown to 10 torr may be varied from approximately 1.5 hours to 7.5 hours depending on the number of rough pumps operating. It is not expected that the insulation will be adversely affected by rate of pressure change within those limits, and, therefore, all five rough pumps will be used, pumping the chamber to 10^{-1} torr in about four hours and to 10^{-4} torr in approximately six to ten hours. LN_2 cooled traps between the chamber and diffusion pumps will be cooled down before going below 400 microns to prevent back-migration of diffusion pump oils into the chamber which could change emissive characteristics of the coatings and/or insulation. LN_2 cooling chamber walls will be turned on as soon as the pressure drops below 10^{-2} torr to cryopump moisture out-gassed from the tanks and super insulation.

B. Solar Simulator

As soon as the LN_2 cooled shroud is cooled down, the solar simulator will be turned on with a programmed on-off cycle to simulate the selected orbit. The albedo-earth I.R. simulator panels will also be turned on at this time. Both radiant energy sources are energized at this time to start to establish the stable shield temperature pattern/cycle as soon as possible, and will remain energized for the duration of the test.

C. Tank Filling

Chamber pressure will be maintained at a level below 1×10^{-4} for a period of time sufficient to thoroughly outgas the insulation at which time the Propellant Storage tanks will be filled with either LN_2 to simulate the semi-cryogenic fluids or with pre-chilled (0^oF)

alcohol to simulate the earth storable propellant. Simultaneously with tank cool-down, the chamber will be back-filled with dry N_2 gas to increase the free convection heat transfer in the insulation. This is done to expedite establishment of a temperature gradient in the insulation that will approximate the expected steady state thermal gradient.

As soon as the required thermal gradient has been established in the insulation, the chamber will be pumped down below 1×10^{-5} torr. One hour after reaching 10^{-5} torr, the LN_2 tank will be topped off and the fill lines sealed. Power will also be applied to the heater on the alcohol tanks to bring the alcohol temperature to a value equal to the predicted average shield temperature.

The helium cryopump system will be kept on standby during chamber pumpdown and will be activated if it appears that the increased pumping speed is necessary to drop chamber pressure below 1×10^{-5} torr.

D. <u>Data Reading</u>

As soon as equilibrium and/or a reproducible temperature distribution and boil-off rate has been established in the tanks and shields, a 48 hour period of data recording will be started. Chamber pressure and solar beam intensity will be recorded continuously and all other parameters such as cold shroud temperature, meteorite shield temperature, alcohol and LN_2 temperatures in the tanks and LN_2 boil-off rates will be recorded at 1/4 to 1/2 hour intervals. At the end of the recording period, the test will be terminated.

E. Shut-down

Chamber shut-down procedure will consist of first shutting down the helium cryopumping system and the flow of LN_2 through the shroud. Diffusion pumps are then turned off while continuing to circulate LN_2 through the traps to prevent back migration of oil while the pump oil cools down. The chamber will then be vented to 500 microns with dry nitrogen gas to assist shroud warm-up by convection heat transfer. The solar simulator and I.R. simulator may also be operated at reduced power during this period to accelerate the warm-up procedure.

A continuous purge of dry nitrogen gas will be bled into the chamber during the shroud warm-up period to assist the rough pump in removing gasses evaporated from the shroud or cryopump panels, and to reduce back migration of rough pump oil into the chamber As soon as the cooled surfaces in the chamber exceed 0°C, the chamber will be vented to atmosphere through a filter to remove air borne dust and dirt.

Draining of both alcohol and LN_2 from the storage tank will be accomplished during the warm-up period so that removal of the tanks and other equipment from the chamber can be started as soon as venting is completed.

A detailed step-by-step test operating procedure will be prepared and issued prior to the start of these tests.

2.3.3 SES Solar System Performance

With the exception of the solar system and helium cooled cryopumps, all sub-systems in the Space Environmental Simulator Laboratory have been in operation for almost two years.

One of the four quadrants of the solar simulator has been operated under thermal vacuum conditions, and performed very well. Intensity scans taken through the beam during this test indicate that intensity throughout the beam is generally within ± 5% of an average value.

Figure 20 is a plot of beam intensity versus radial position taken through the center of the quadrant. This data was taken in air prior to the vacuum test and is closely duplicated under thermal vacuum conditions.

This and other data indicates that the beam maintains 95% of average intensity on a 16' - 17' diameter depending on location. Intensity data between 0 and 2' radius is not plotted since operation of all four quadrants is required to maintain uniform intensity in this area. Re-alignment of the mosaic mirrors based on a more refined computer program will reduce the variation to within ± 5% limits.

Based on this data taken during operation of the one quadrant, it appears that the Space Environment Simulator will produce a 17.5' diameter beam at $145^{W}/ft.^2$ intensity, with uniformity of ± 5% and collimated within ± 1.5 - 3.5°. Installation of the other three primary collimator quadrants is in progress, and is scheduled to be completed the week of November 24, 1963. As of this date, installation of the equipment is on schedule. Upon completion of the remaining three quadrants of the primary collimator, the chamber will be pumped down to 10^{-5} torr or below, the LN_2 cooled shroud will be cooled to 100°K or below and a series of scans made of the complete solar simulator beam to verify beam size, uniformity, and intensity. This pumpdown cycle will also be used to verify proper performance of all vacuum chamber equipment including the helium cooled

cryopumps and to calibrate the chamber ionization gages against a pair of calibrated ion gages used as secondary standards.

2.4 Malta Testing

Planning for the Malta tests has been initiated and tankage requirements have been factored into the design and procurement schedule. Specifically, three (3) two-foot diameter double-walled spheres, complete with meteorite shields, will be individually tested in a cryogenically cooled high-vacuum chamber. By using three tanks, the problem of using a fuel followed by an oxidizer is eliminated. Also, the tank rework requirements for testing a cryogenic fluid followed by an earth storable fluid are minimized.

The tank/fluid combinations and sequence are as follows:

Tank #1 - Run #1 - LN_2

Tank #1 - Run #2 - OF_2

Tank #3 - Run #3 - B_2H_6

Tank #2 - Run #4 Methyl Alcohol

Tank #2 - Run #5 Aerozene 50

Tank #1- Run #6 N_2O_4

The present plan for reproducing the solar simulation of the SES chamber at Malta is to install heating pads on the meteorite shield and establish the thermal distribution obtained in the SES pumpdowns.

Work Planned for October

3.1 Thermal Analysis

Analyses will be performed relative to vehicle orbit cycle, performance prediction of tests, and piping heat flow.

3.2 Structural Design and Hardware Procurement

The major effort in October will be concerned with following the fabrication and testing of the tankage, support systems, and meteorite shields. A significant amount of vendor liaison will be necessary to insure design compliance. An assembly procedure will be prepared to detail the method of assembly of tanks, shields, and support systems. This will document the steps that should be taken prior to, and after tank insulation.

3.3 Test Preparation and Operation

During October the design, fabrication, and ordering of test support equipment should be essentially completed.

3.4 Malta Testing

Several tasks will be initiated in October to insure that the facility will be ready. The basic test plan and procedures will be drafted. Second, a definition of the required boil-off metering equipment and valves will be made. Finally, the design of the propellant transfer system and the OF_2 liquefication equipment will commence.

4.0 REFERENCES

1. Armour Research Foundation Report No. ARF 3207-14, 1962.
2. Black, I.A. and Glaser, P.E.: "The Performance ot a Double-Guarded Cold Plate Thermal Conductivity Apparatus", A. D. Little Report presented at the 1963 Cryogenic Engineering Conference.
3. Journal of Research of the National Bureau of Standards -- Vol. 57, No. 6, Research Paper No. 2726, December 1956.

Table 1

Comparison of Support Systems' Thermal Performance

Support System	Heat Leak (Btu/hr.)	Boil-off Rates (% per year)
Hard Mount	3720	1467
Tension Members (Rods)	68.32	26.9
Washer Supports	217	85.4
Active Supports	6.26	2.5

List of Figures

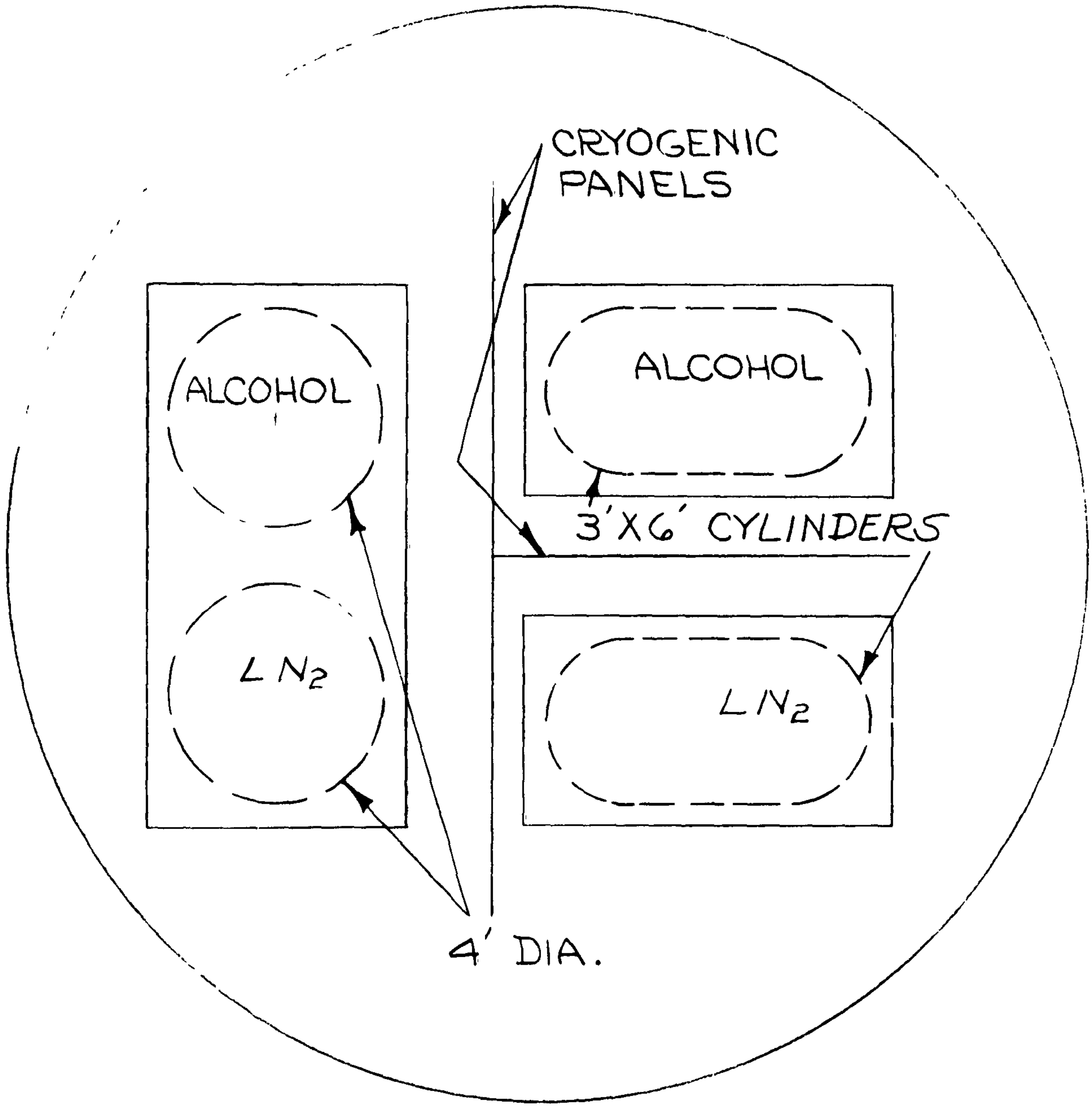


FIGURE 1

FIRST PUMPDOWN IN SES

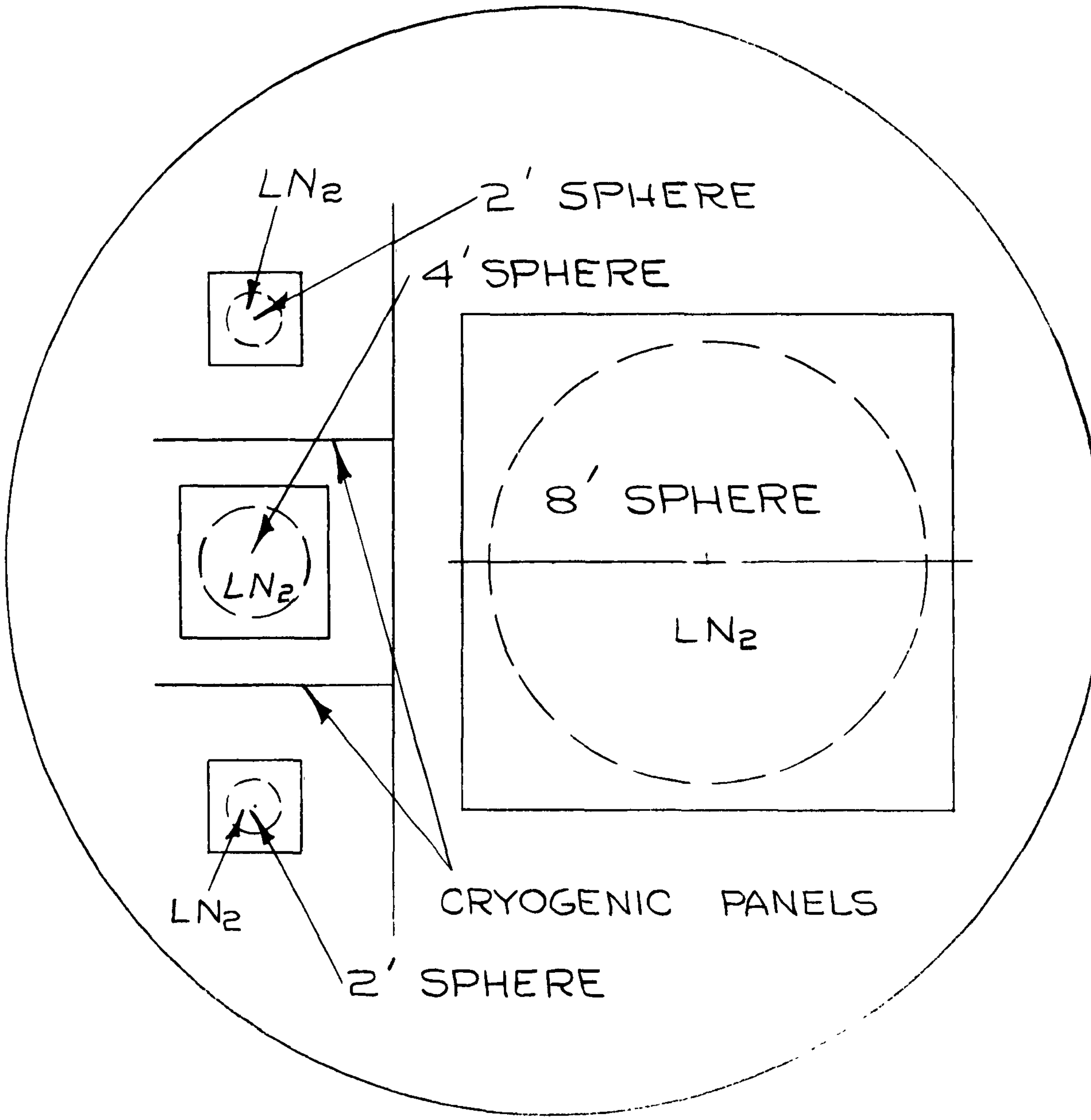


FIGURE 2

SECOND PUMPDOWN IN SES

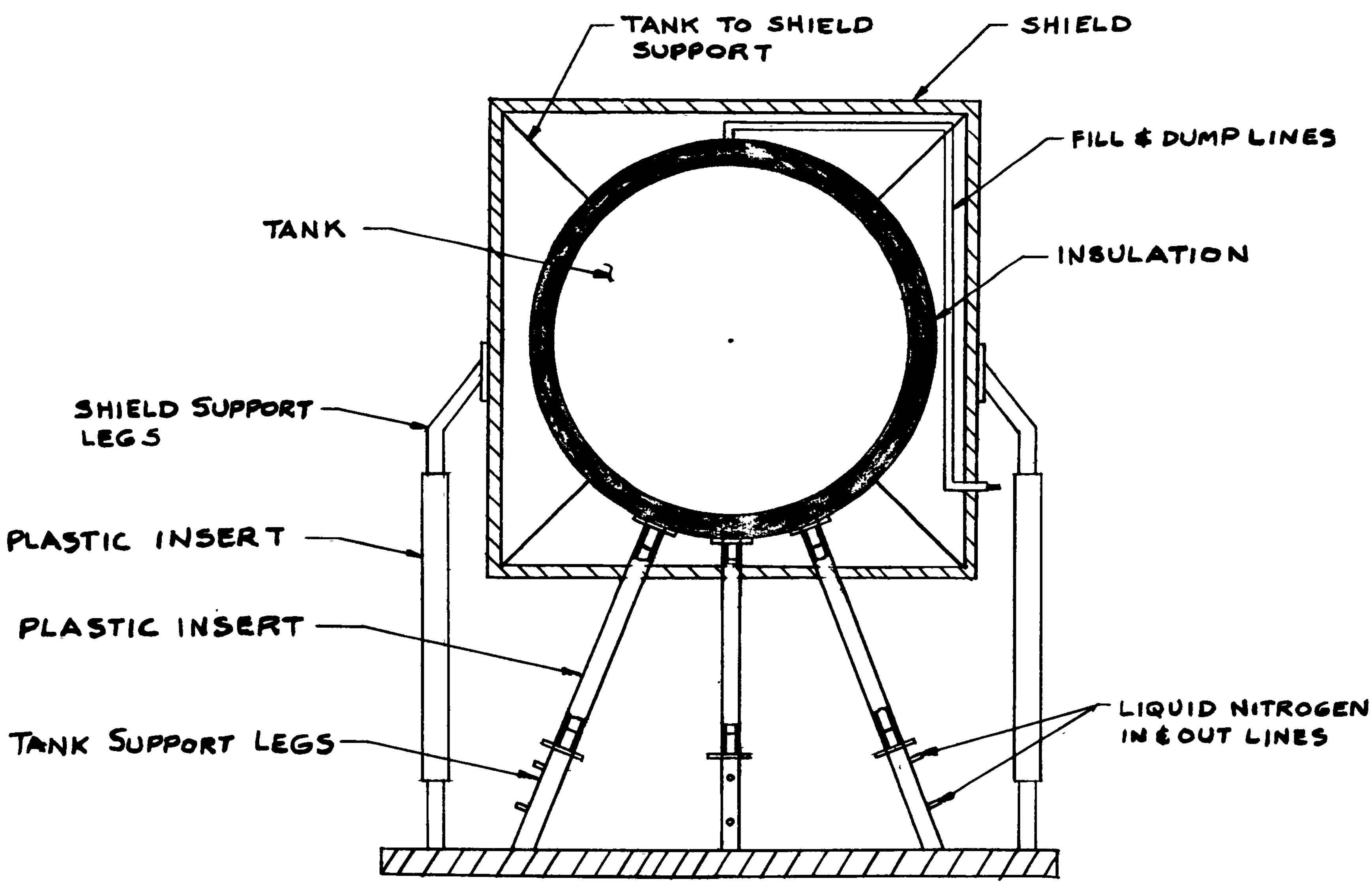
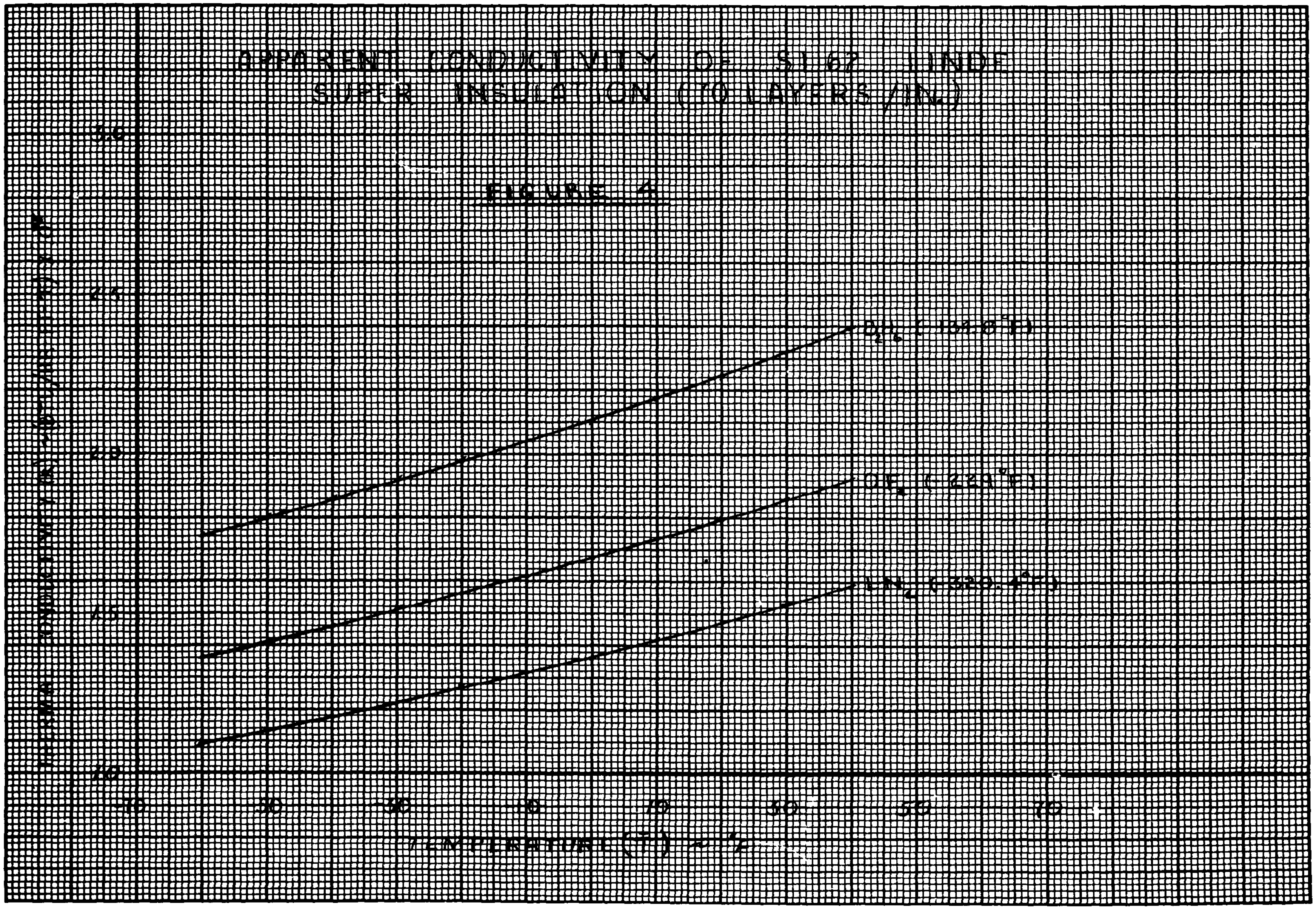


FIG. 3

SCHEMATIC OF TANK, SHIELD AND SUPPORT SYSTEM



APPARENT CONDUCTIVITY OF S1-62 LINDE
SUPER INSULATION (70 LAYERS/IN.)
FIGURE 4
TEMPERATURE (T) ~ °F

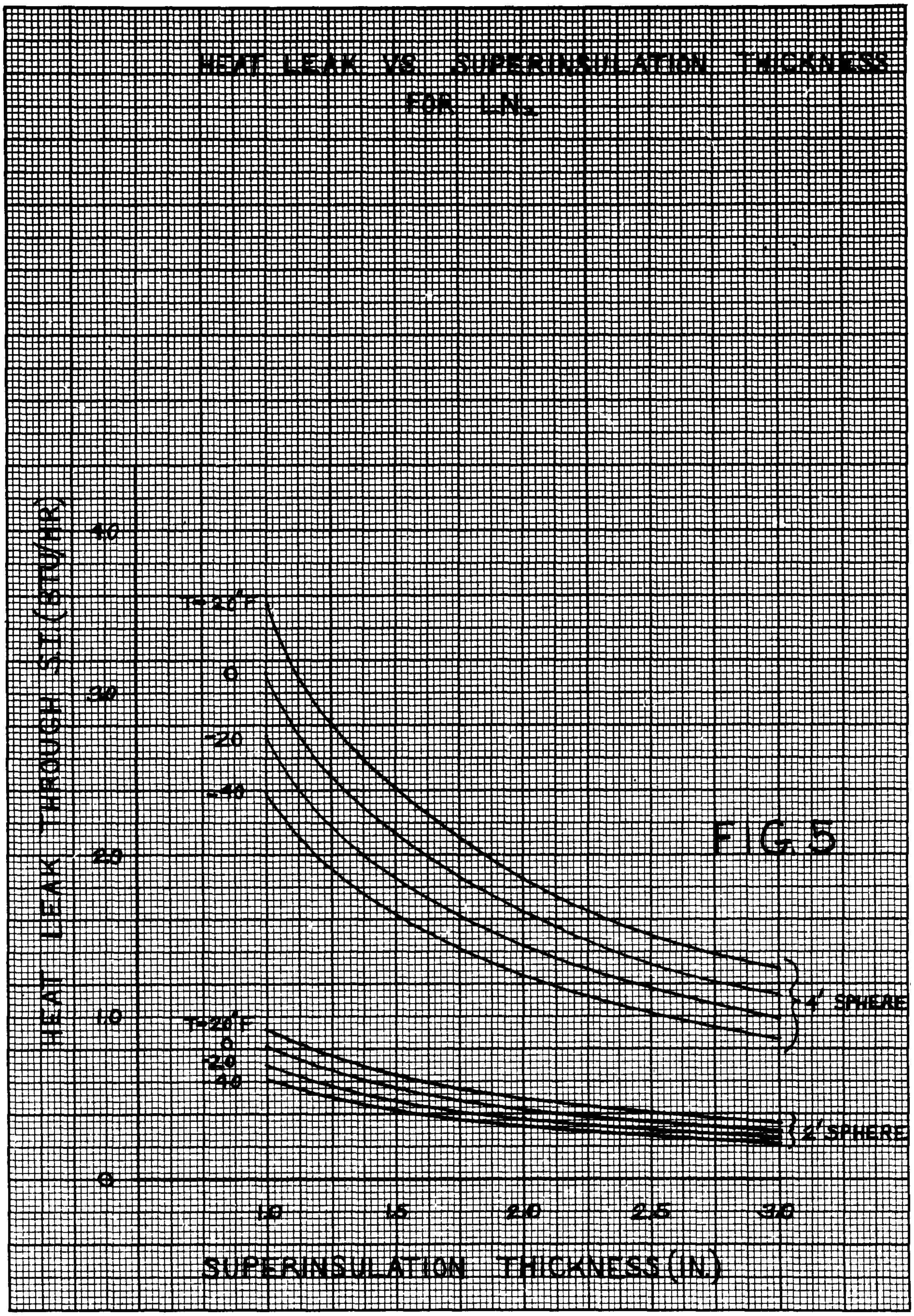

HEAT LEAK VS SUPERINSULATION THICKNESS
FOR LN2
HEAT LEAK THROUGH S.T. (BTU/HR)
40
30
20
10
0
T=20°F
0
-20
-40
T=20°F
0
-20
-40
4' SPHERE
2' SPHERE
1.0
1.5
2.0
2.5
3.0
SUPERINSULATION THICKNESS (IN.)

FIG. 5

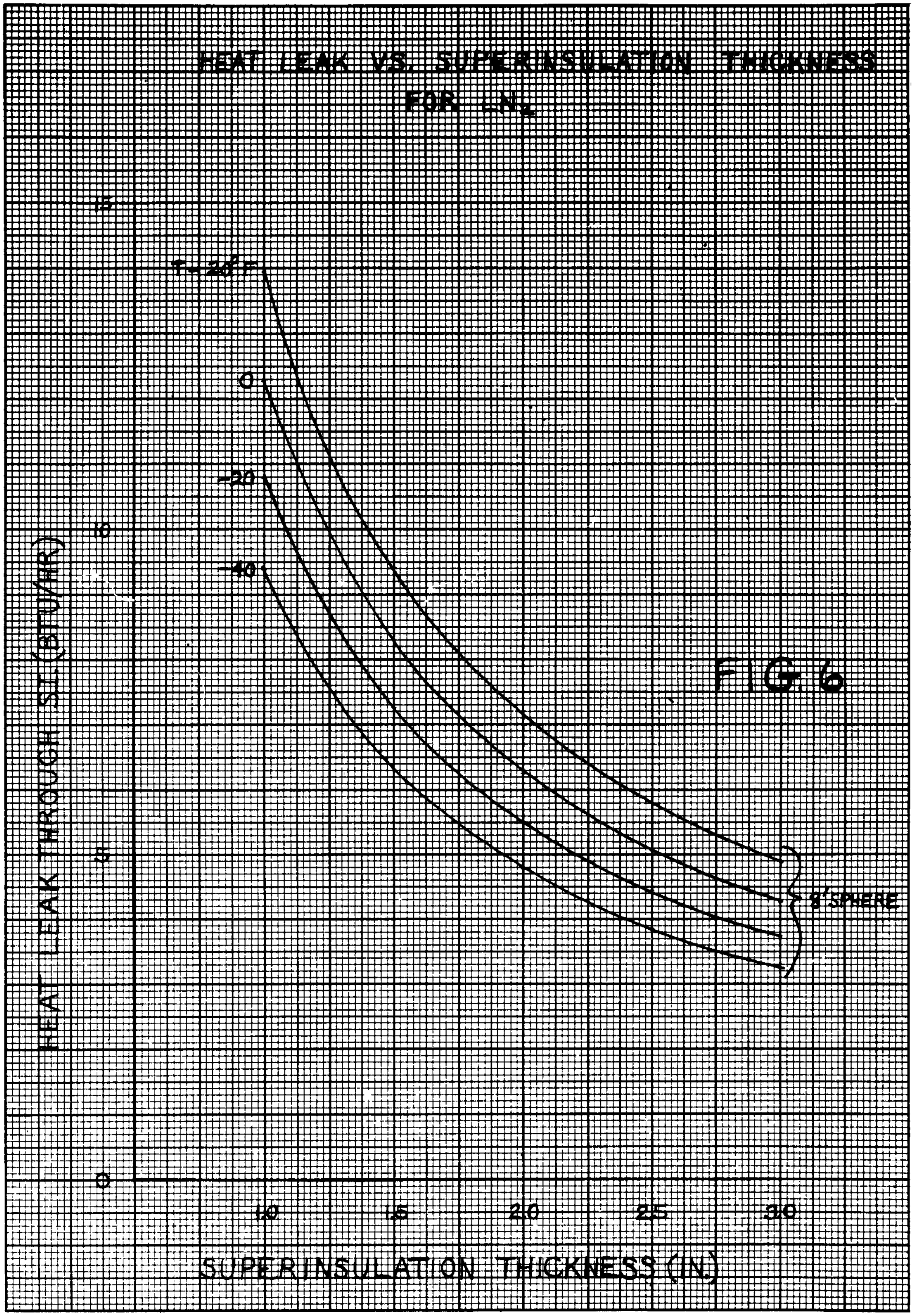
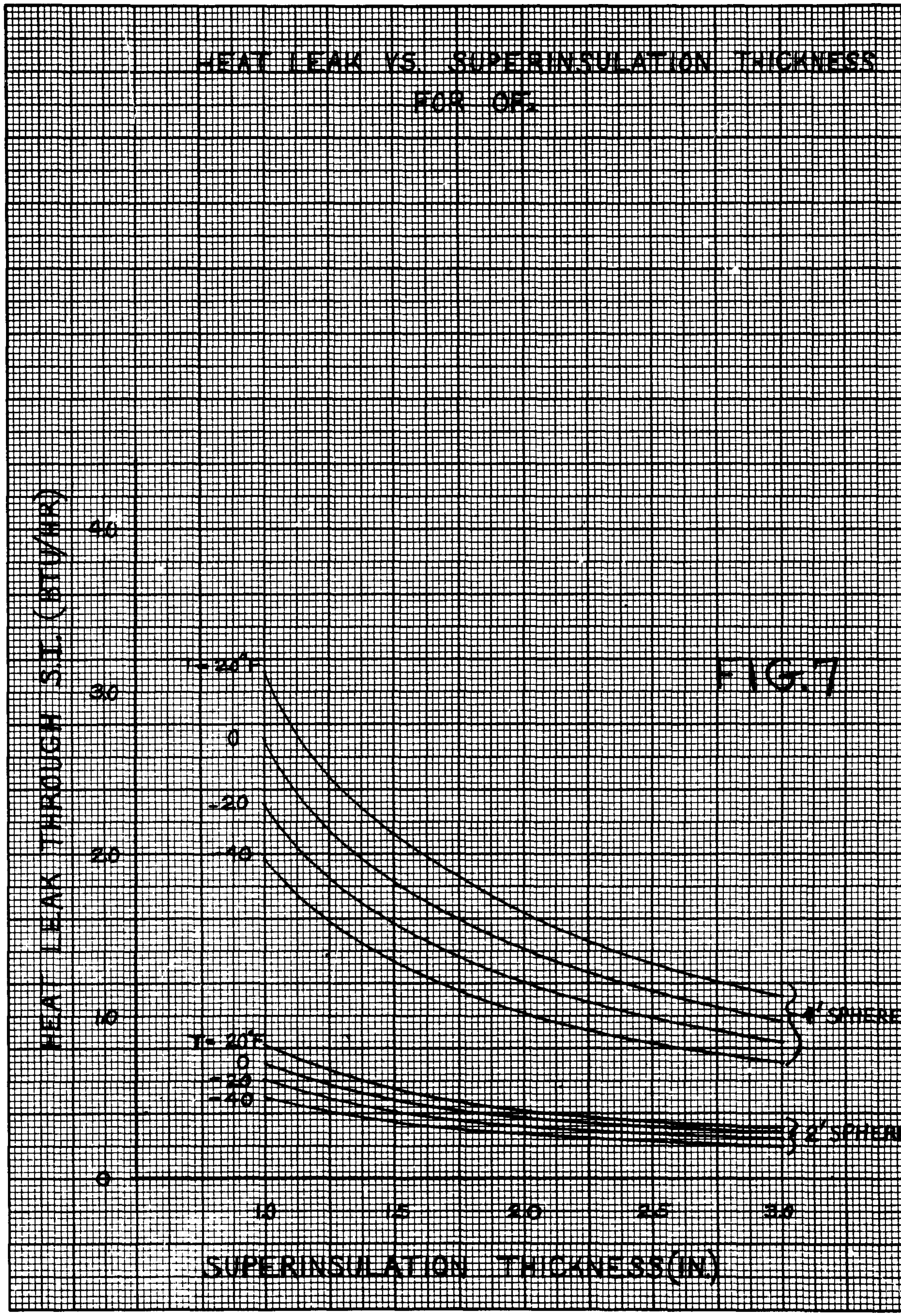

HEAT LEAK VS. SUPERINSULATION THICKNESS
FOR LN_2
HEAT LEAK THROUGH S.I. (BTU/HR)
15
10
5
0
T=20°F
0
-20
-40
8' SPHERE
1.0
1.5
2.0
2.5
3.0
SUPERINSULATION THICKNESS (IN.)

FIG. 6



HEAT LEAK VS. SUPERINSULATION THICKNESS
FOR OF2
HEAT LEAK THROUGH S.I. (BTU/HR)
40
30
20
10
0
T=20°F
0
-20
-40
T=20°F
0
-20
-40
4' SPHERE
2' SPHERE
1.0
1.5
2.0
2.5
3.0
SUPERINSULATION THICKNESS (IN.)

FIG. 7

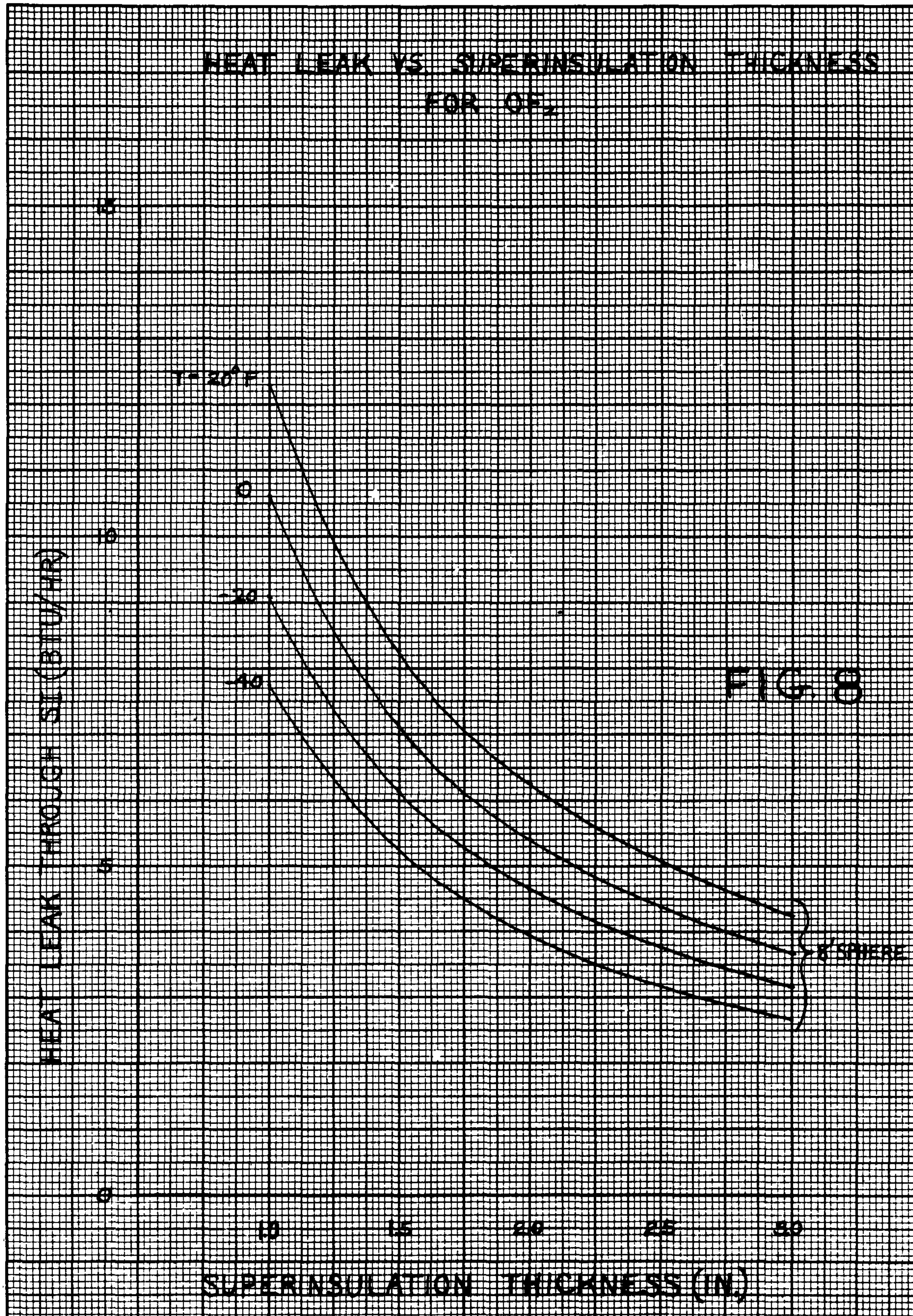

HEAT LEAK VS. SUPERINSULATION THICKNESS
FOR OF_2
15
10
5
0
HEAT LEAK THROUGH S.I. (BTU/HR)
T= 20° F
0
-20
-40
8' SPHERE
1.0
1.5
2.0
2.5
3.0
SUPERINSULATION THICKNESS (IN.)

FIG. 8

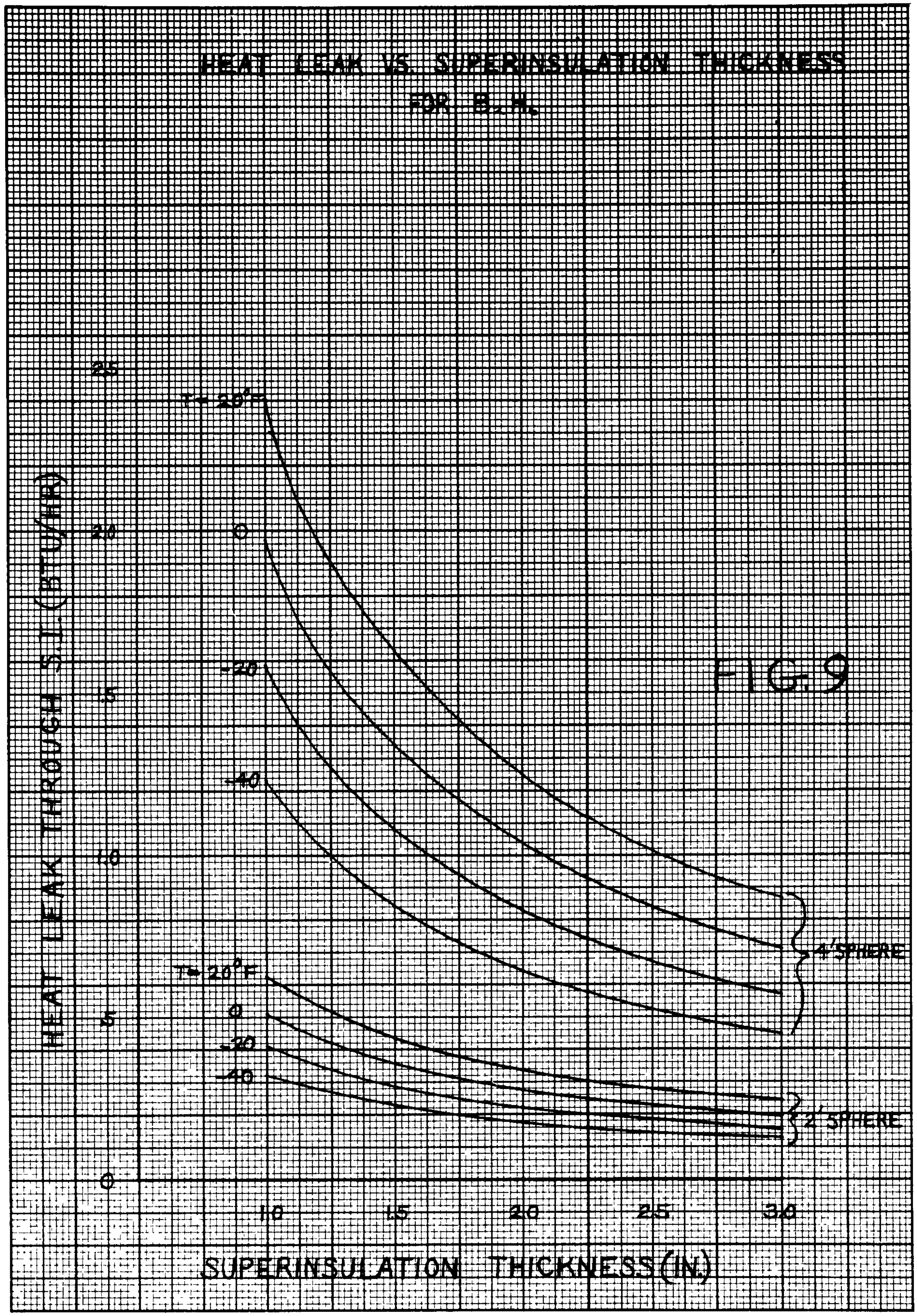

HEAT LEAK VS. SUPERINSULATION THICKNESS
FOR B. H.
HEAT LEAK THROUGH S.I. (BTU/HR)
25
20
15
10
5
0
T=20°F
0
-20
-40
T=20°F
0
-20
-40
4' SPHERE
2' SPHERE
1.0
1.5
2.0
2.5
3.0
SUPERINSULATION THICKNESS (IN.)

FIG. 9

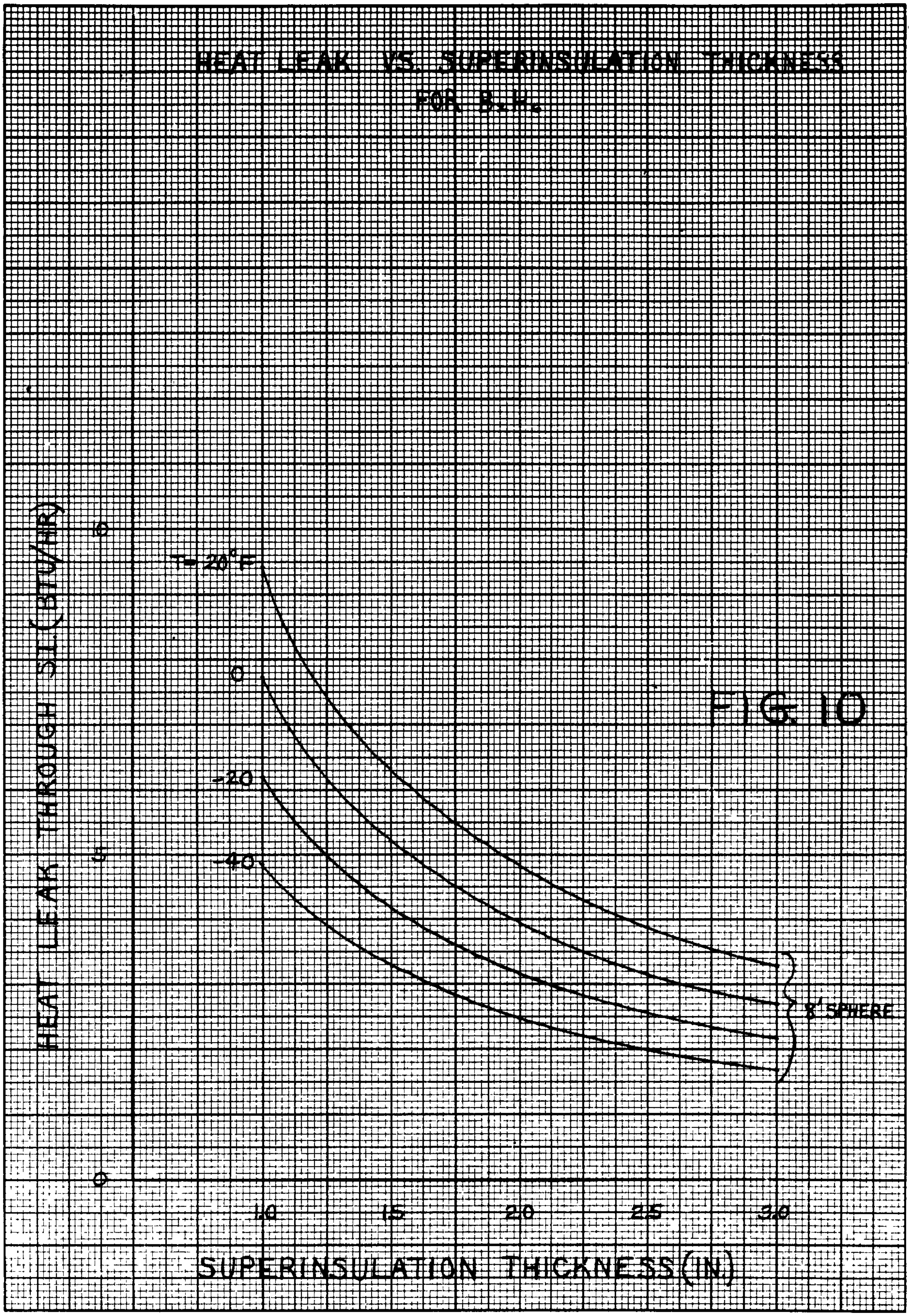

HEAT LEAK VS. SUPERINSULATION THICKNESS
FOR B.H.
HEAT LEAK THROUGH SI (BTU/HR)
10
5
0
T=20°F
0
-20
-40
8' SPHERE
1.0
1.5
2.0
2.5
3.0
SUPERINSULATION THICKNESS (IN)

FIG. 10

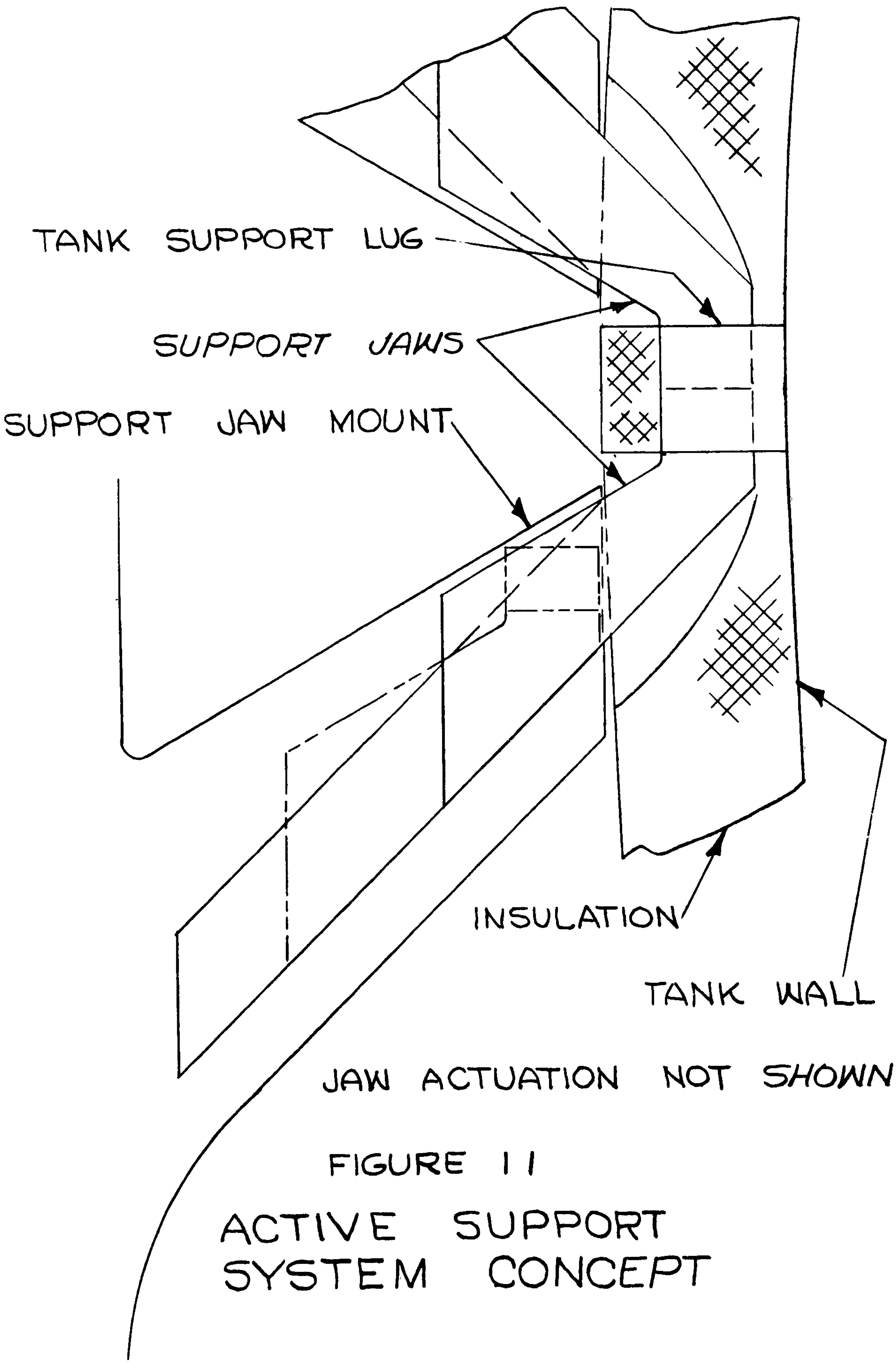


FIGURE 11

ACTIVE SUPPORT
SYSTEM CONCEPT

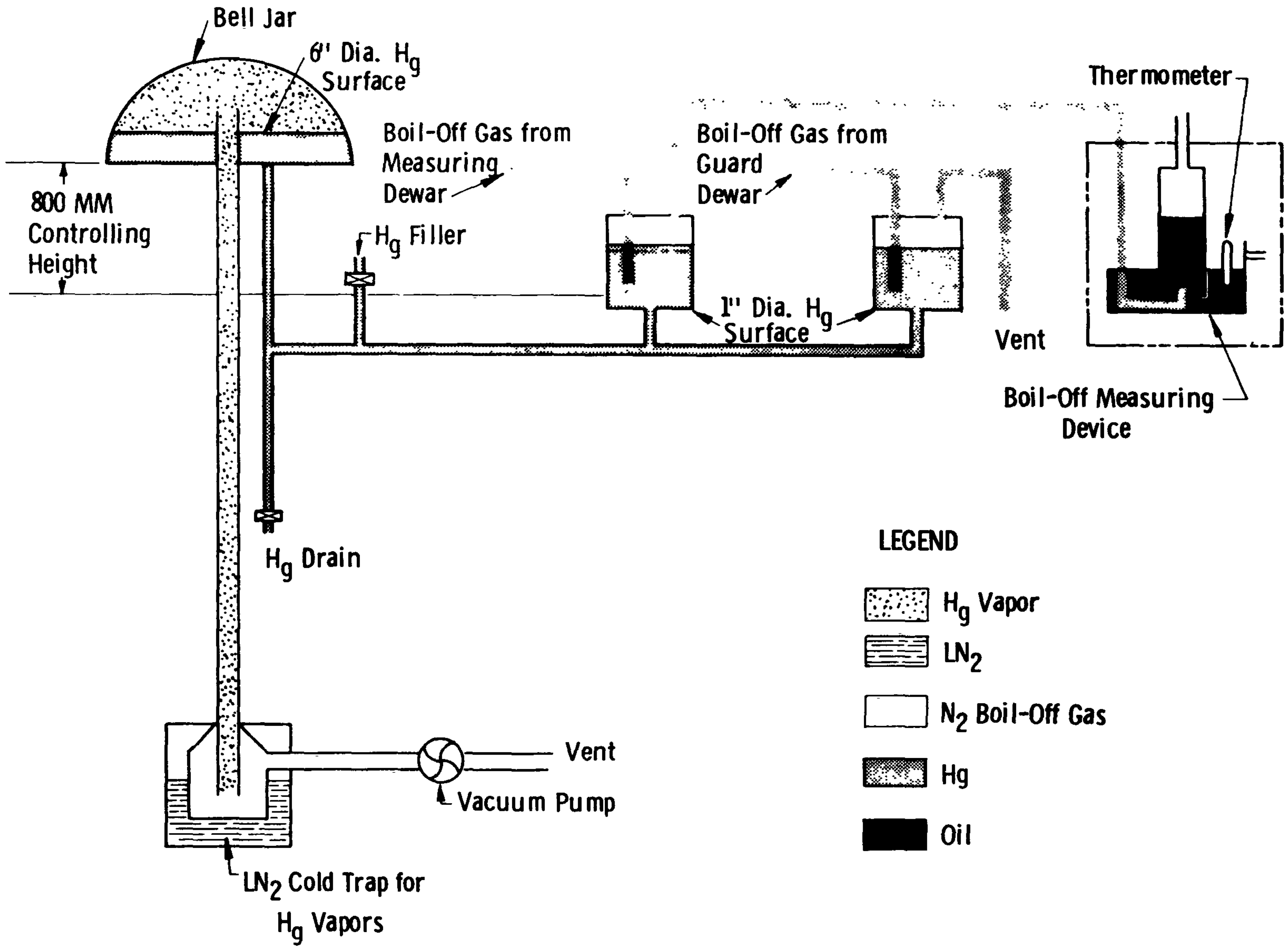
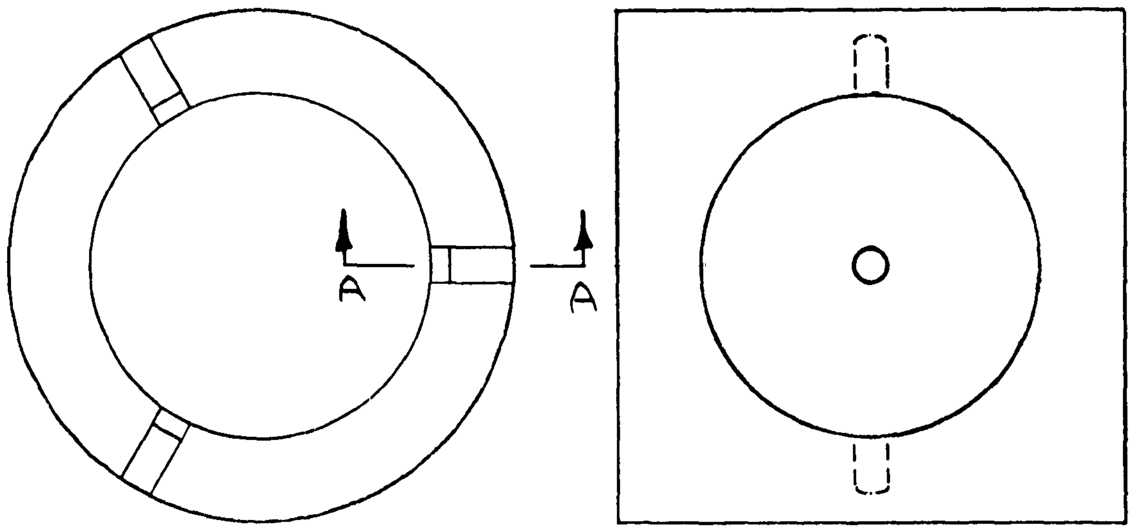
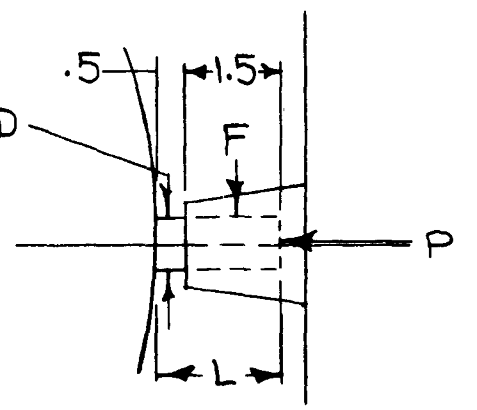


FIGURE 12 DIAGRAM OF GAS PRESSURE CONTROL SYSTEM (Reference 2)





SECTION A-A

FIGURE 13
HARD MOUNT

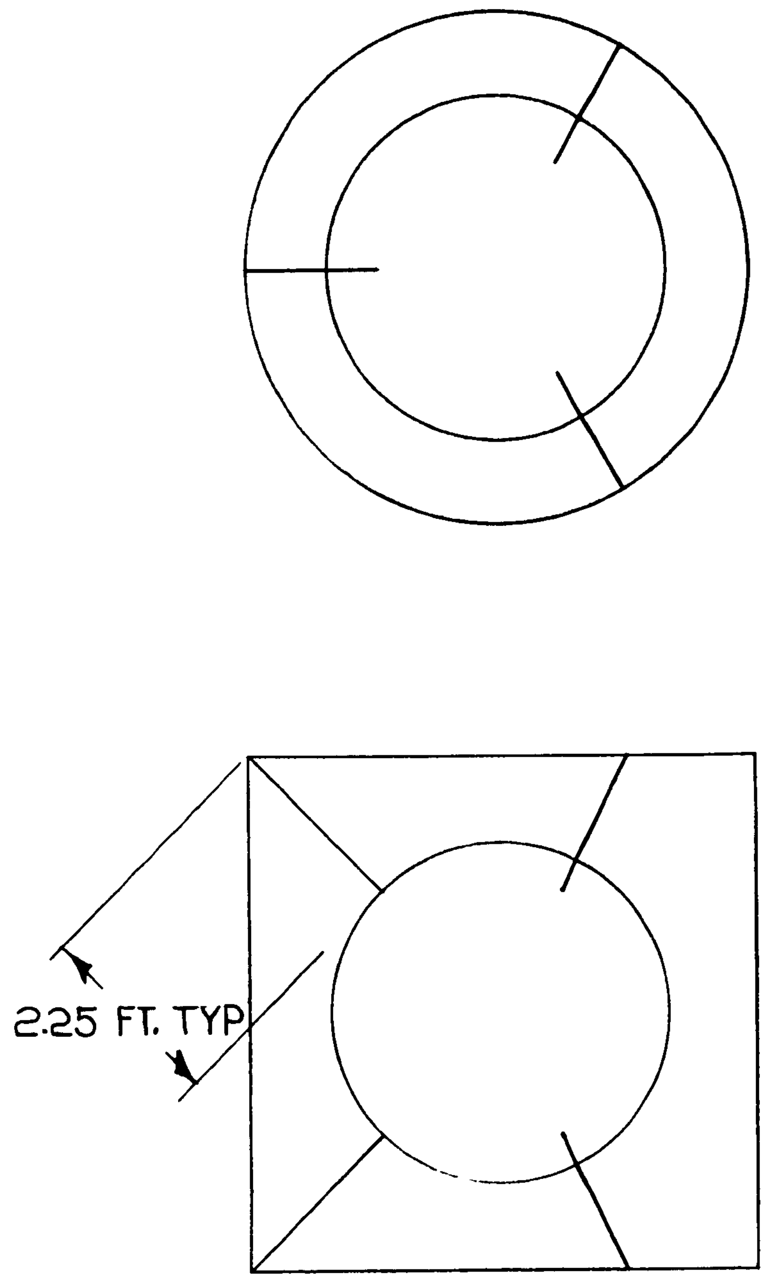


FIGURE 14
TENSION SUPPORTS

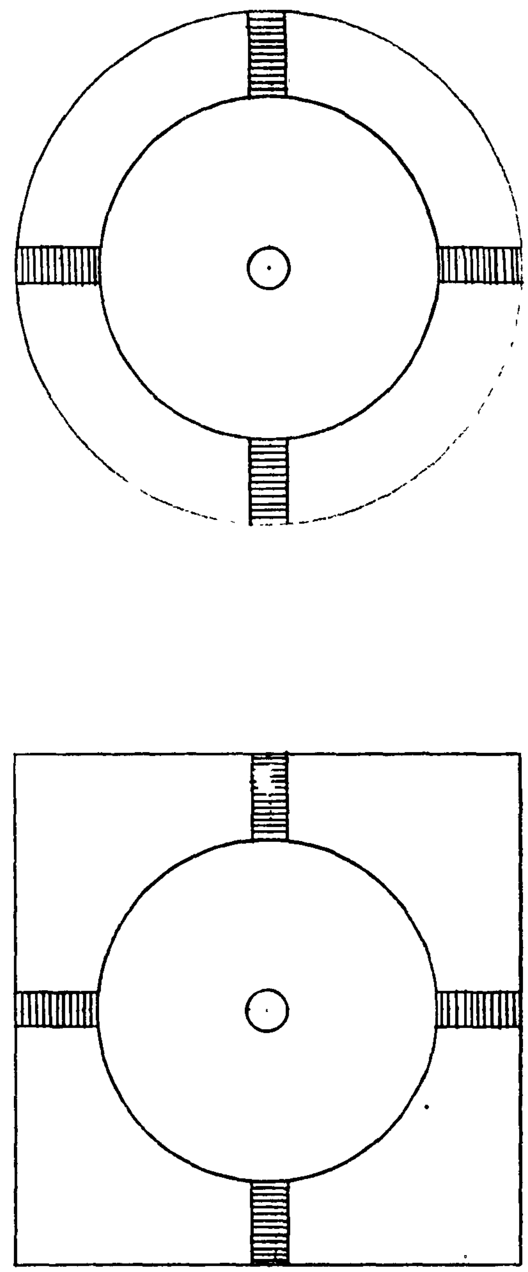
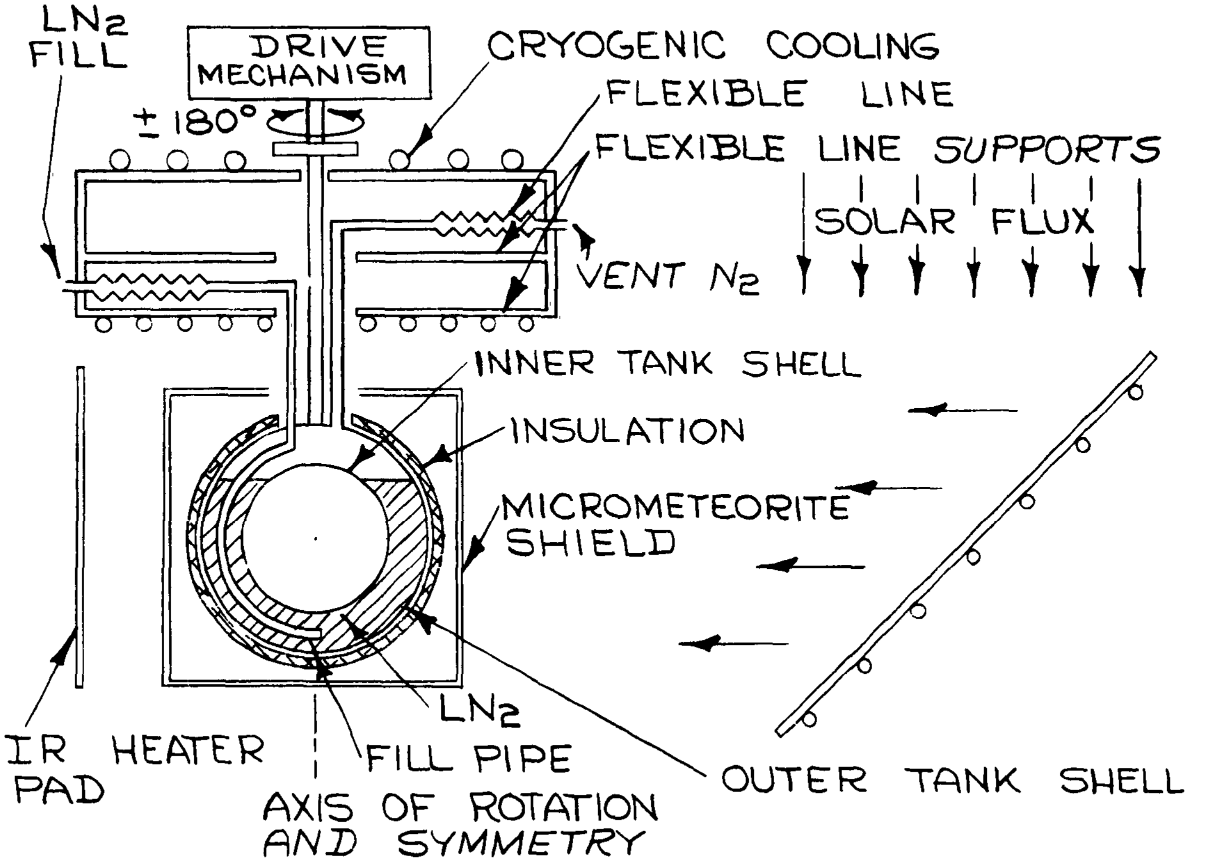


FIGURE 15
COMPRESSED WASHER SUPPORTS



GEOMETRIC RELATIONSHIP OF FIXED AND ROTATING TANKS TO THE SOLAR FLUX

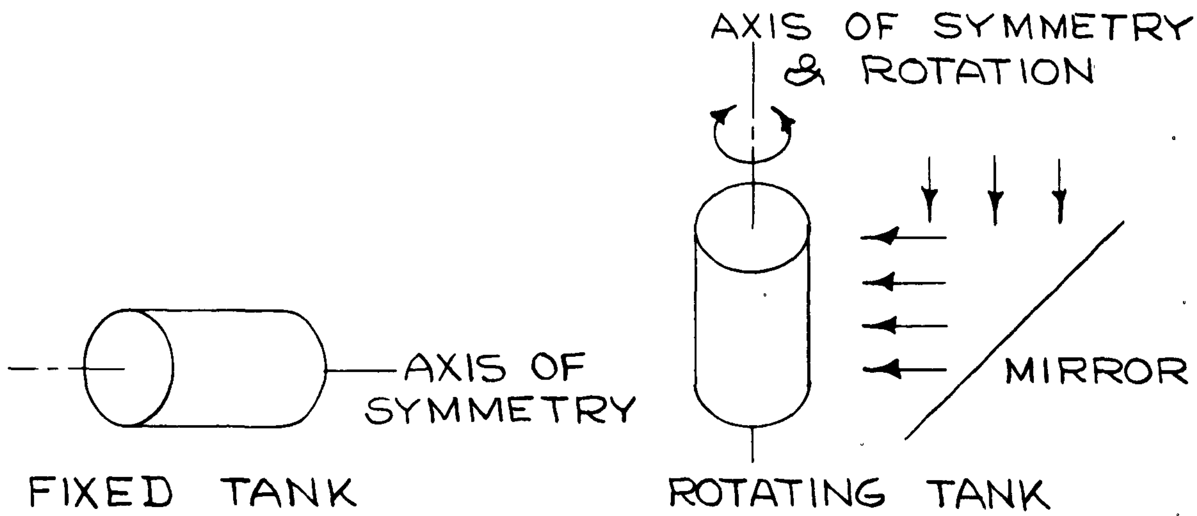


FIGURE 16

SIMPLIFIED DIAGRAM OF THE 2′ DIA ROTATING TANK EXPERIMENT

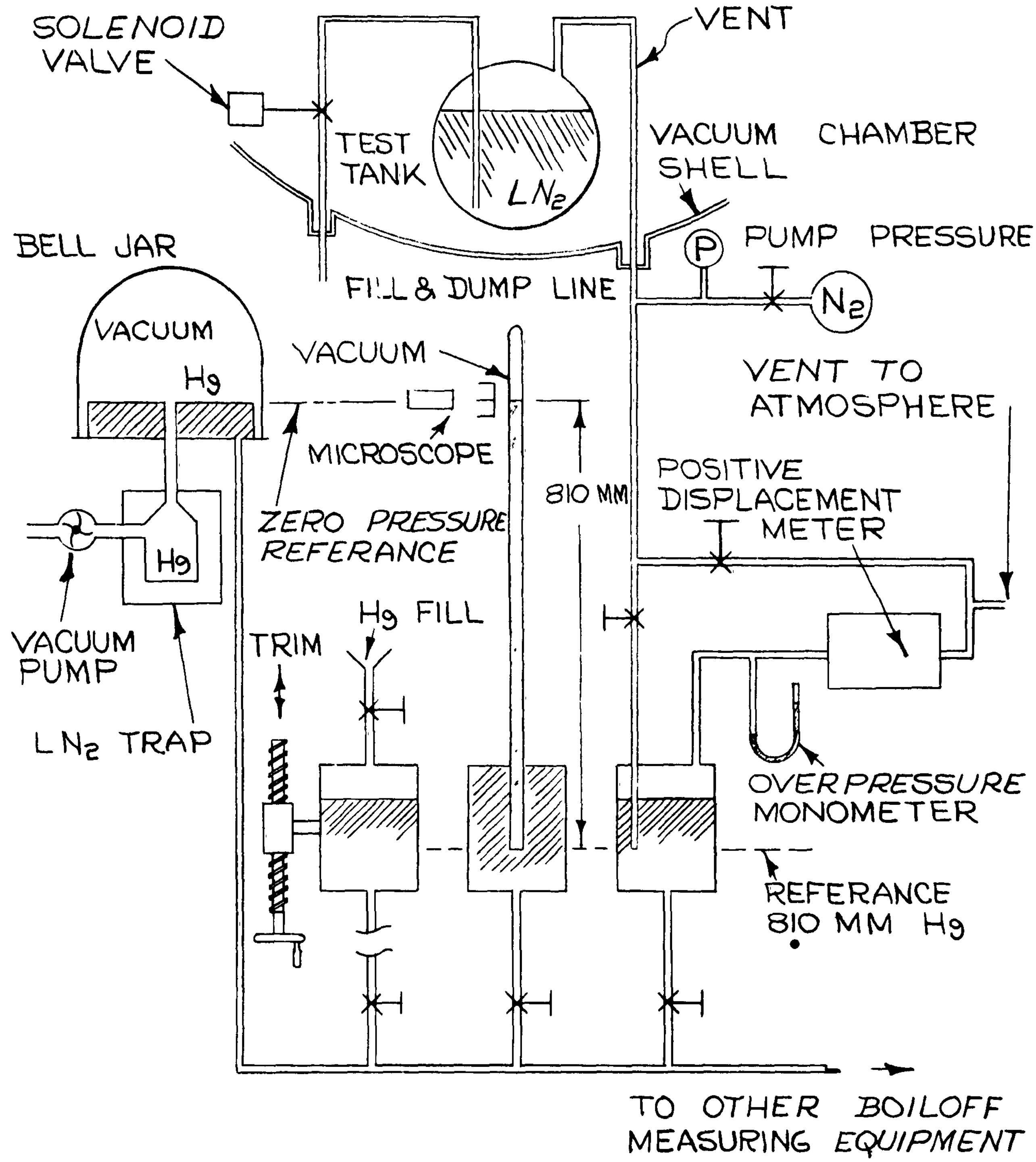


FIGURE 17

SIMPLIFIED DIAGRAM OF THE LN_2 TEST TANK BOILOFF MEASUREMENT SYSTEM

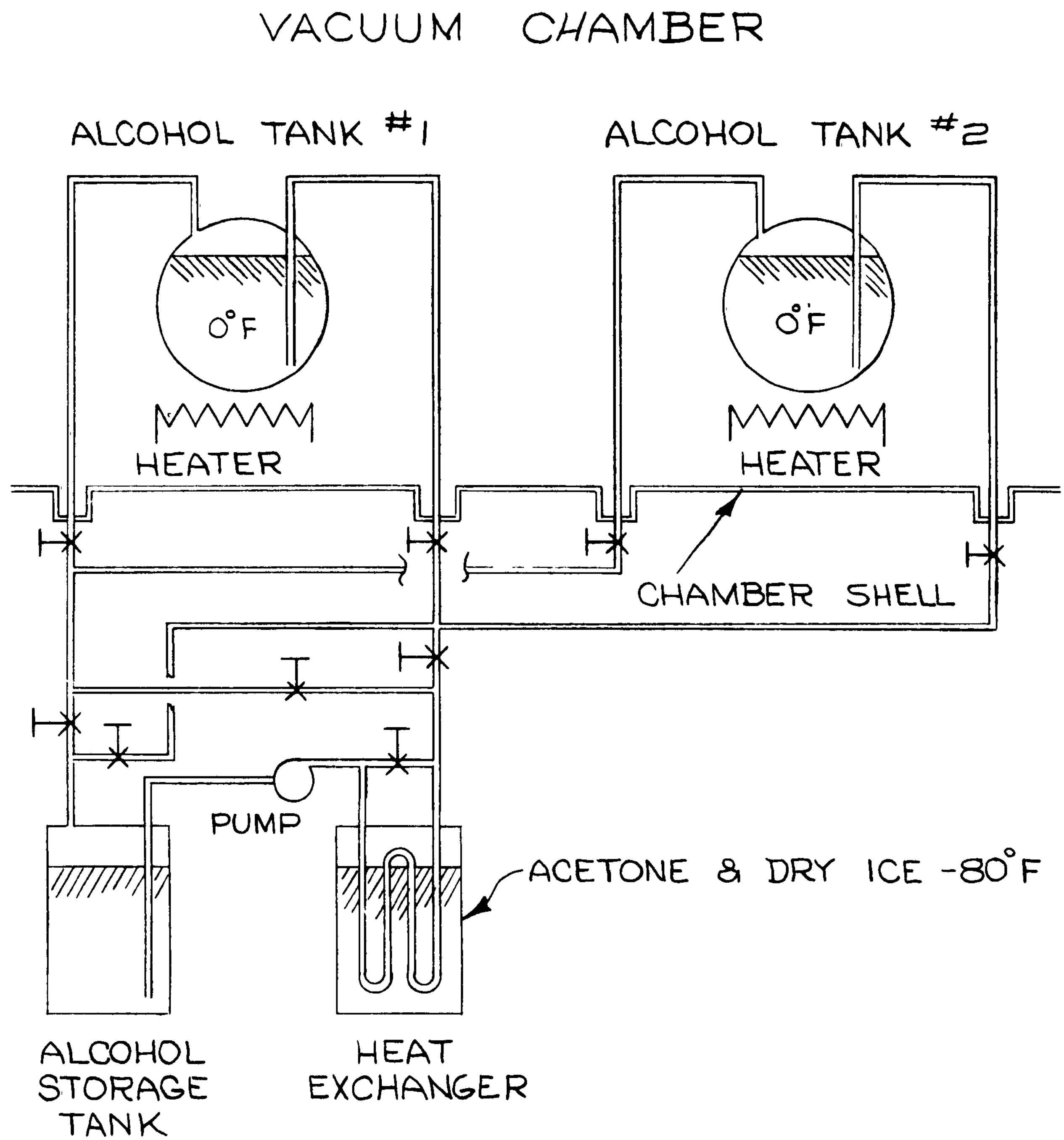
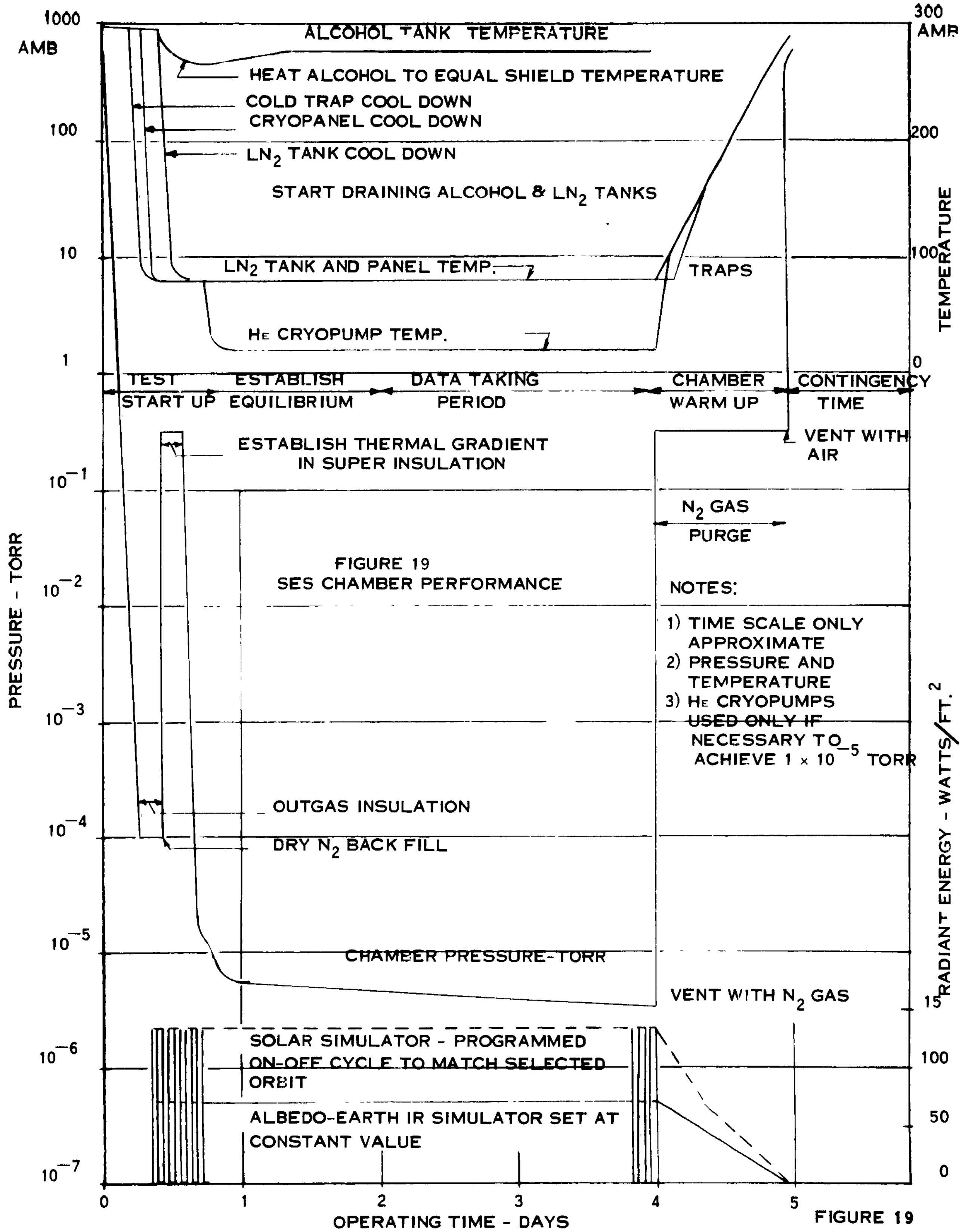


FIGURE 18

SIMPLIFIED PIPING DIAGRAM
OF THE
ALCOHOL TANK FILL AND DRAIN SYSTEM



1000
AMB
100
10
1
10^{-1}
10^{-2}
10^{-3}
10^{-4}
10^{-5}
10^{-6}
10^{-7}
PRESSURE - TORR
300
AMB
200
100
0
TEMPERATURE
ALCOHOL TANK TEMPERATURE
HEAT ALCOHOL TO EQUAL SHIELD TEMPERATURE
COLD TRAP COOL DOWN
CRYOPANEL COOL DOWN
LN_2 TANK COOL DOWN
START DRAINING ALCOHOL & LN_2 TANKS
LN_2 TANK AND PANEL TEMP.
TRAPS
He CRYOPUMP TEMP.
TEST START UP
ESTABLISH EQUILIBRIUM
DATA TAKING PERIOD
CHAMBER WARM UP
CONTINGENCY TIME
ESTABLISH THERMAL GRADIENT IN SUPER INSULATION
VENT WITH AIR
N_2 GAS PURGE
FIGURE 19
SES CHAMBER PERFORMANCE
NOTES:
1) TIME SCALE ONLY APPROXIMATE
2) PRESSURE AND TEMPERATURE
3) He CRYOPUMPS ~~USED ONLY IF~~ NECESSARY TO ACHIEVE 1×10^{-5} TORR
OUTGAS INSULATION
DRY N_2 BACK FILL
CHAMBER PRESSURE-TORR
VENT WITH N_2 GAS
SOLAR SIMULATOR - PROGRAMMED ON-OFF CYCLE TO MATCH SELECTED ORBIT
ALBEDO-EARTH IR SIMULATOR SET AT CONSTANT VALUE
150
100
50
0
RADIANT ENERGY - WATTS/FT.^{2}
0
1
2
3
4
5
OPERATING TIME - DAYS

FIGURE 19

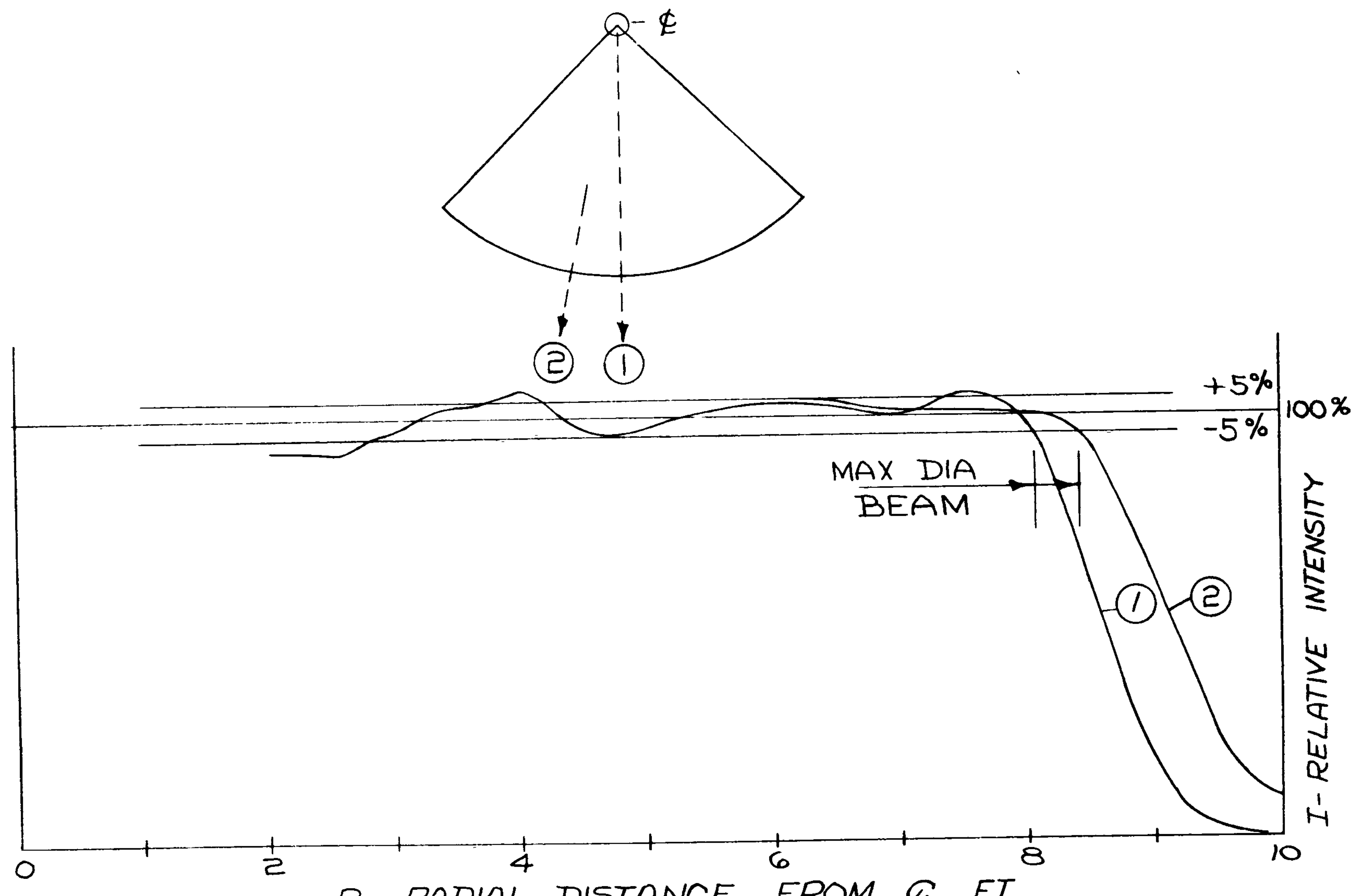


FIGURE 20

SPACE ENVIRONMENT SIMULATOR
SOLAR SYSTEM PERFORMANCE